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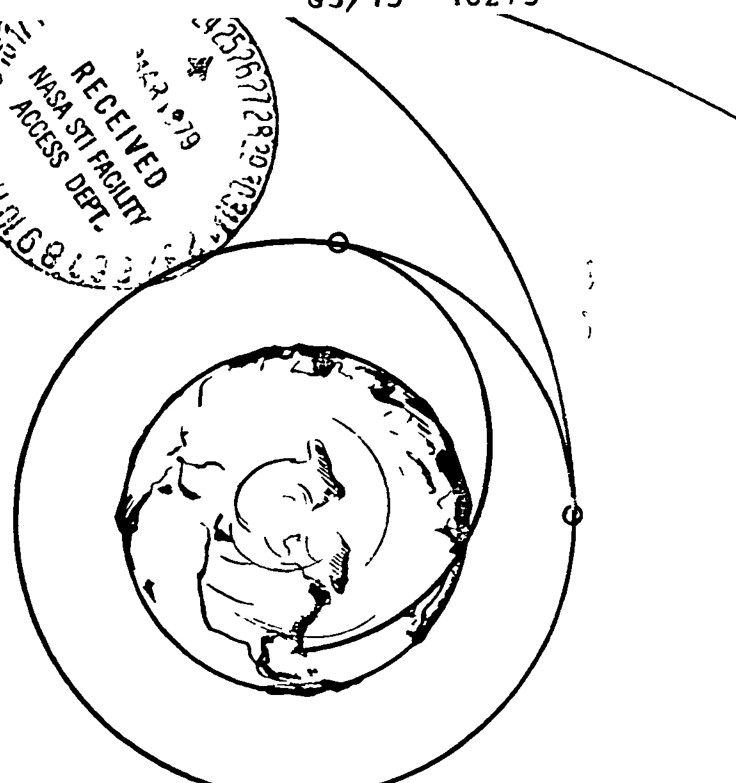

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EARTH ORBITAL ASSESSMENT OF SOLAR ELECTRIC
AND SOLAR SAIL PROPULSION SYSTEMS

to

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

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TABLE OF CONTENTS

TABLE OF CONTENTS
(Continued)

LIST OF APPENDICES

LIST OF TABLES

TABLE OF CONTENTS
(Continued)

LIST OF FIGURES

ABSTRACT

The Earth orbital applications potential of Solar Electric (Ion Drive) and Solar Sail low-thrust propulsion systems are evaluated. Emphasis is placed on mission applications in the 1980s. The two low-thrust systems are compared with each other and with two chemical propulsion Shuttle upper stages (the IUS and SSUS) expected to be available in the 1980s The results indicate limited Earth orbital application potential for the low-thrust systems in the 1980s (primarily due to cost disadvantages) The longer term potential is viewed as more promising. Of the two systems, the Ion Drive exhibits better performance and appears to have better overall application potential.

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INTRODUCTION

Two alternative low-thrust propulsion systems are being considered for development and use on a possible rendezvous with Comet Halley in 1986. Both systems would make use of solar radiation to satisfy the high propulsive energy requirements associated with the mission.

One of the candidates is the Solar Electric Propulsion (SEP) or Ion Drive system depicted in Figure 1. Large arrays of solar cells with concentrators would concentrate and convert sunlight into electrical energy to operate mercury ion thrusters. The other candidate propulsion system is the Solar Sail; the current baseline design, the Heliogyro, is shown in Figure 2. The Heliogyro consists of a dozen ultrathin (0.1 mil), very long (7.5 km) blades mounted on a hub. The Heliogyro rotates about the hub axis and is propelled through space by the pressure of solar radiation incident on its blades.

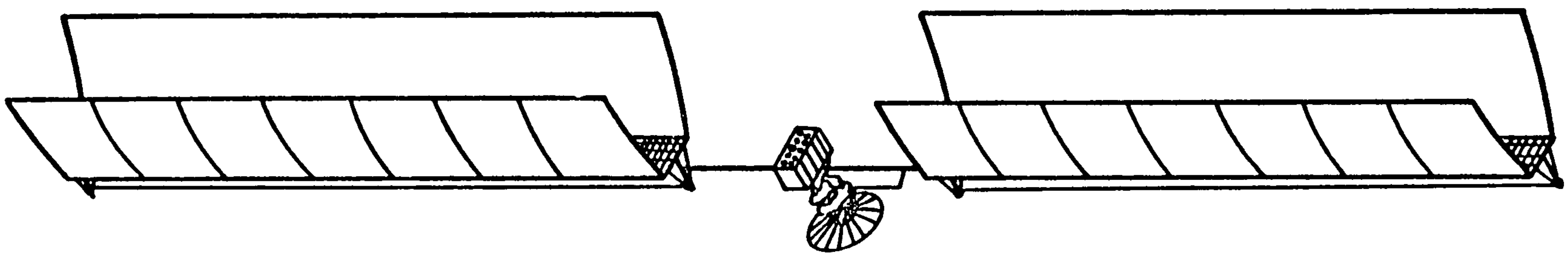


FIGURE 1. HALLEY MISSION ION DRIVE SYSTEM[1]*

*Superscript numbers refer to references presented at the end of the text.

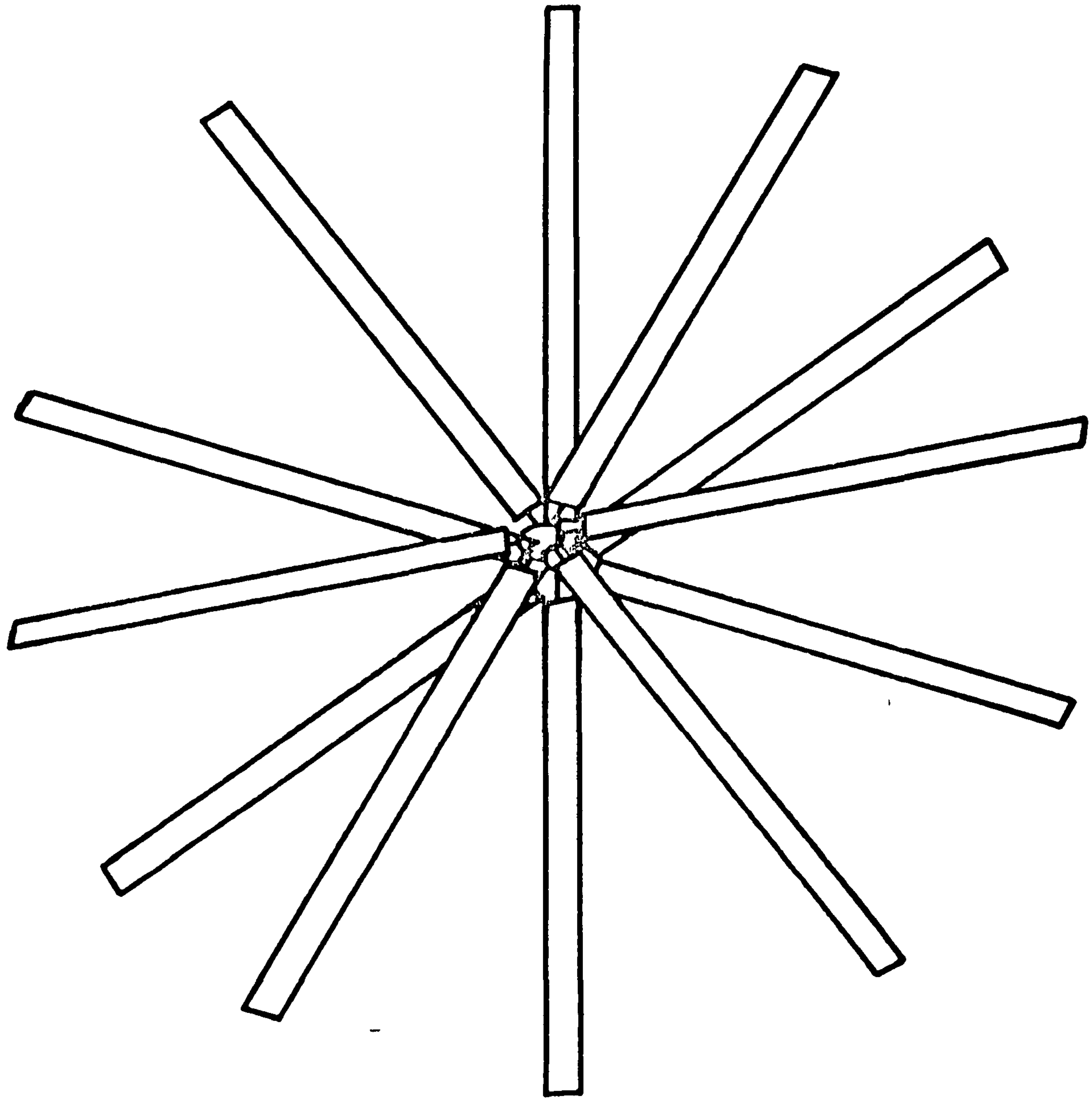


FIGURE 2. HALLEY MISSION SOLAR SAIL (HELIOGYRO) SYSTEM [2]

If either of these propulsion systems were developed for the Halley mission it would be available for use on a number of other automated planetary-type missions and also for Earth orbital applications. This report is an examination of the Earth orbital applications potential of the two competing systems. Emphasis is placed on system performance in those applications expected to materialize by the mid-1980s, when the Halley system first would become available for use.

SUMMARY

There are a variety of promising future Earth orbital applications of low-thrust propulsion. These include payload delivery, payload servicing, technology verification (e.g., space solar power generation experiments), orbit debris control, and manned mission support. However, in the decade of the 1980s, the use of low-thrust systems in Earth orbit would primarily be limited to geosynchronous payload delivery, and perhaps some involvement in a space solar power generation program, if one develops. Other mission opportunities are not likely to arise prior to the 1990s.

Use of low-thrust systems in the payload delivery role would require that a propulsion module developed for a planetary application (e.g., Halley's rendezvous) be developed further into a fully autonomous stage. Guidance, navigation, communications, control and other capabilities (e.g., rendezvous and docking) not included in the module would have to be added.

The performance of Ion Drive and Solar Sail systems in a geosynchronous delivery role was evaluated. Because of the existence of the Van Allen radiation belts and the susceptibility of solar cells to radiation damage, two mission profiles were considered for the Ion Drive system. Profile "A" assumed that a reusable Ion Drive system is stationed in a park orbit, called an "exchange" orbit, of 15,000 to 16,000 km altitude, which is above the most damaging region of the radiation belts. An IUS was assumed to be used to transfer payloads from the Shuttle in low orbit to the Ion Drive in the exchange orbit. The Ion Drive would provide transportation from the exchange orbit to geosynchronous orbit. Profile B assumed that the Ion Drive would operate directly from the Shuttle cargo bay Solar arrays were assumed to be oversized so as to minimize the effects of power loss due to damage in the radiation belts. This profile eliminated the need for IUS support, but it significantly increased flight time and reduced reuse possibilities. Profile A flight times were determined to be 60 to 110 days, while those for Profile B were over 200 days.

Because the Sail could not operate directly from the Shuttle at low orbit altitudes (aerodyanmic drag forces would overcome solar pressure), only one mission profile was considered for the Solar Sail. The Sail's minimum operational altitude (for circular orbits) was estimated to be in the 1000 to 1500-km range. A small two-stage solid rocket motor system (derived from the IUS) was assumed to be used to transfer payloads from the Shuttle to the Sail. The Sail would be used

to transport payloads from a 1000 to 1500-km exchange orbit to geosynchronous orbit. This would require travel through the radiation belts, where the Sail material would be weakened by radiation exposure. The damage incurred would not degrade performance (as does Ion Drive power loss), but could affect Sail lifetime.

Low-thrust segments of the Ion Drive Profile B and the Sail profile are similar and were used as the basis for a comparison of the performance of the two systems. For the Ion Drive system, flight time to geosynchronous orbit was determined as a function of payload for a wide range of payloads. The data were generated with a rapid performance analysis technique developed for that purpose. The Sail's unique thrust pointing constraints limited the extent of Sail trajectory analysis that could be accomplished with available analysis tools (or those tools that could be developed or obtained within the term of the study). Nonetheless, the general level of expected Sail performance was estimated from data generated, and compared to the more definitive results obtained for the Ion Drive system. The comparison indicated that the Ion Drive would produce better Earth orbital performance (shorter flight times) than the Solar Sail.

The sensitivity of low-thrust performance to system parameter degradation (e.g., system weight growth) was evaluated. Sensitivity was found to be similar for the two low-thrust systems and less than that for chemical systems. For chemical systems, degradation of system parameters is more likely to mean loss of capability to perform its intended mission.

The extent to which low-thrust propulsion systems might be used in the 1980s to deliver payloads to geosynchronous orbit was assessed by comparing low-thrust transportation costs to those of competing chemical systems. Transportation costs were analyzed for two sets of payloads representing a projected level of geosynchronous mission traffic and a range of payload definitions. The use of multiple payload stacks was assumed for both chemical and low-thrust systems. Larger stacks were assumed for the low-thrust system so as to take advantage of its greater performance and reduce the required number of trips. However, the results indicated that low-thrust payload delivery would cost more than chemical system delivery. The high cost of low-thrust stage hardware and services (∿$30 million for an Ion Drive or Sail stage versus ∿$5 million for the IUS) could not be offset by reasonable increases in payload stack size, even though the low-thrust system was assumed to be reused several times.

It was concluded that significant potential exists for the application of low-thrust systems to Earth orbital missions. However, in the 1980s, the most likely application is the delivery of payloads to geosynchronous orbit. Currently defined low-thrust propulsion systems are not competitive with chemical systems for the delivery of the single or multiple small automated payloads expected to dominate geosynchronous mission traffic in the 1980s. On the other hand, if a requirement develops for delivery of large single payloads (e g , space solar power system elements, or large space antennas), then the use of low-thrust propulsion may prove desirable.

Although Solar Sail performance estimates must be regarded as preliminary, it appears that the Ion Drive system would produce better performance and would have better overall application potential for Earth orbital missions.

MODIFICATIONS FOR EARTH ORBITAL APPLICATIONS

As developed for the Halley mission application, the Ion Drive and Solar Sail systems would be propulsion modules dependent upon the Halley spacecraft for guidance, navigation, and communications functions. In addition, the Solar Sail would derive electrical power from radioactive thermoelectric generators (RTGs) located on the spacecraft and the Ion Drive could require auxiliary control from the spacecraft reaction control system (RCS). This integrated spacecraft/propulsion module design approach is practical when both systems must be expended (as on the Halley mission), and/or propulsion module costs are small compared to total mission costs, and/or unique benefits can be derived by integrating the two (e.g., Ion Drive solar arrays might be used to satisfy large spacecraft power requirements at the mission destination).

For Earth orbital applications the integrated spacecraft/propulsion module might be satisfactory for some future missions involving large relatively high-cost spacecraft with large power requirements. However, if low-thrust propulsion is to become a useful system for a wide variety of Earth orbital applications, then the module must evolve into a fully autonomous and probably reusable stage. Thus, guidance, navigation, communications, control, and other capabilities (e.g., rendezvous and docking) must be added. In addition, some design modifications will result from the fact that the Earth orbital operating environment is significantly different from that of the Halley and planetary missions. Some conditions unique to Earth orbit are:

(1) The Van Allen radiation belts which surround the Earth contain high-energy particles that damage solar cells and degrade the strength of sail material.

(2) Many Earth orbital trajectories periodically pass through the Earth's shadow, producing thermal cycling and on-off power and propulsion cycling.

(3) Solar intensity is approximately constant (except during shadowing) at the 1 a.u. value placing fewer demands on low-thrust system design and operation.

(4) At low orbital altitudes aerodynamic drag becomes strong enough to dominate and restrict operation of large-area low-thrust systems.

(5) The possibility of collision with other manmade objects in Earth orbit may be a significant hazard for systems employing large-area structures such as the Ion Drive solar arrays and particularly the Sail blades.

(6) Thrust vector steering ranges and turning rates are much larger for Earth orbits because the vehicle direction of motion with respect to the Sun is constantly and rapidly changing.

The conversion of low-thrust modules to stages will require additional development effort. For the Ion Drive, the required development was defined and costed in a 1975 Boeing study of a 25-kw Solar Electric Propulsion Stage (SEPS).[3] Based upon the Boeing results as adjusted for inflation and the larger Ion Drive solar arrays, a revised estimate of module-to-stage developmental conversion costing $27 million FY 1977 dollars was generated. In the case of the Sail, no detailed study of the conversion has yet been made. The cost of Sail conversion was estimated to be slightly higher than that of Ion Drive, particularly if a rendezvous and docking capability is required. Due to the Sail's limited maneuverability and extreme structural flexibility, the rendezvous and docking problem is expected to be more difficult to solve than for the Ion Drive. In addition, a separate power source (e.g., small solar cell arrays, batteries, or RTGs) not required for the Ion Drive would have to be provided Considering these factors, a preliminary estimate of the Sail conversion cost of at least $28-30 million is projected

MISSION DEFINITION

A variety of potential future Earth orbital applications have been developed in previous studies of low-thrust propulsion. The 1976 Boeing "Payload Utilization of SEPS (PLUS)" study[4] dealt exclusively with Earth orbital missions. Mission concepts studied included: payload delivery, geosynchronous orbit space servicing, technology verification, orbit debris removal, and manned mission support. These concepts are summarized in the following paragraphs.

Payload Delivery

Payload delivery is the only concept supported in the near future by currently planned missions. Low-thrust stage performance greatly exceeds that of existing and currently planned high-thrust chemical systems. This increase in performance can produce potential cost savings, e.g., by making possible the transportation of large multiple payload stacks from low orbit to geosynchronous orbit. However, the economic viability of low thrust in the mission role is dependent upon the capabilities, costs, and characteristics of not only the low-thrust system but also those of the Shuttle, the payloads and the Interim Upper Stage (IUS) Low-thrust systems could prove definitely more desirable in a payload delivery role if larger single payloads evolve that require not only the additional propulsion capability but perhaps also need low accelerations to prevent structural damage (e.g , large, flexible antennas).

Servicing

The PLUS study concluded that up to 40 percent savings in program costs could be realized by instituting SEPS-based servicing operations in geosynchronous orbit. The servicing system, illustrated in Figure 3, consists of three hardware elements: the SEPS, a geosynchronous parts or replacement module warehouse, and an automated servicer with a manipulator arm to effect transfer of replacement modules to a spacecraft. When not in use, the servicer would remain docked to the warehouse When servicing is required, the SEPS would dock with the servicer and the servicer would transfer needed replacement modules to its own storage bays. The SEPS would take the servicer to the spacecraft and dock with the spacecraft; the servicer would then make the necessary module replacements. Once servicing is completed, the SEPS would return the servicer to the warehouse, and then the SEPS would be free for spacecraft orbit transfer missions or other functions.

Implementation of the geosynchronous servicing concept requires that all participating spacecraft be of a new low-cost, lower-reliability, modular design type. Overall program reliability would be maintained at a high level through the servicing operations. The projected 40 percent program cost reduction would be achieved through savings in transportation costs (fewer spacecraft taken to orbit) and spacecraft production costs (fewer spacecraft, low-cost designs).

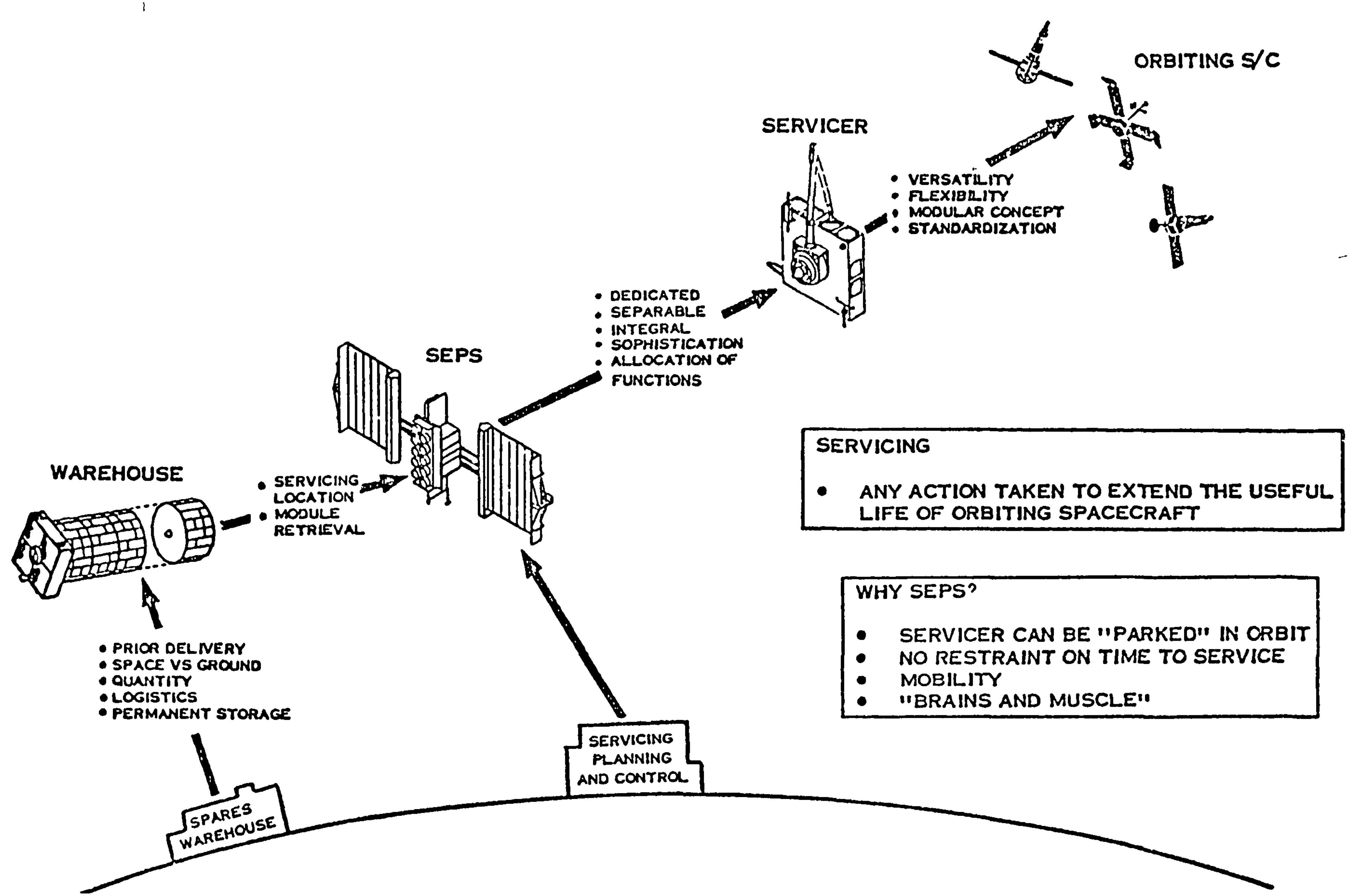


FIGURE 3. GEOSYNCHRONOUS SERVICING CONCEPT(4)

Success of the geosynchronous servicing concept will depend on (among other things) the cooperation and involvement of spacecraft designers and users. This cooperation is not likely to be forthcoming until a prototype system has been built and successfully demonstrated, and the economic benefits have been proven to be real. This factor, coupled with the lead times associated with the servicing system development and new spacecraft development, means that evolution from the present mode of geosynchronous operations to one based on space servicing is unlikely to occur prior to the 1990s, even if a decision to proceed with servicing system development were to occur in the early 1980s.

Technology Verification

Technology verification applications could include demonstrations of space solar power generation and power transmission to Earth. Low-thrust propulsion could be used to transport a subscale space power generation station from an assembly point in low orbit to geosynchronous orbit. Multiple sets of the Ion Drive solar arrays could provide the power source for the test.

Orbit Debris Removal

In the longer-term future it is anticipated that very large structures such as automated space solar power stations (SSPSs) and/or manned space stations will be erected in geosynchronous orbit. As of January 1977, there were nearly 200 objects in geosynchronous orbit (spacecraft, spent stages, separation debris, etc.). Through the 1980s this number will increase by several hundred. While these small objects represent relatively little hazard to each other, they may represent a significant hazard to anything the size of an SSPS (~11 km across) or a manned station. Therefore, at some future date, it may become necessary to begin removing some of the objects that are accumulating in geosynchronous orbit. A long-lived low-thrust stage could serve as a host vehicle and propulsion system for a debris collection device such as that depicted in Figure 4.

Manned Space Operations Support

If a Manned Space Station is developed, a low-thrust propulsion system could become an adjunct to it. The station could serve as a base from which a low-thrust system operates to perform many of the roles previously described.

The presence of the station might result in some of those roles being modified or combined For example, servicing operations might be accomplished by workers at the station. The on-orbit free-flying warehouse and automated servicer described previously would not be needed, the low-thrust system would just retrieve the malfunctioning spacecraft and transport it to the station for repairs. Orbit debris removal could be modified to a salvaging program. Inactive spacecraft could be transported to the station, where salvagable parts would be removed for use in the servicing program.

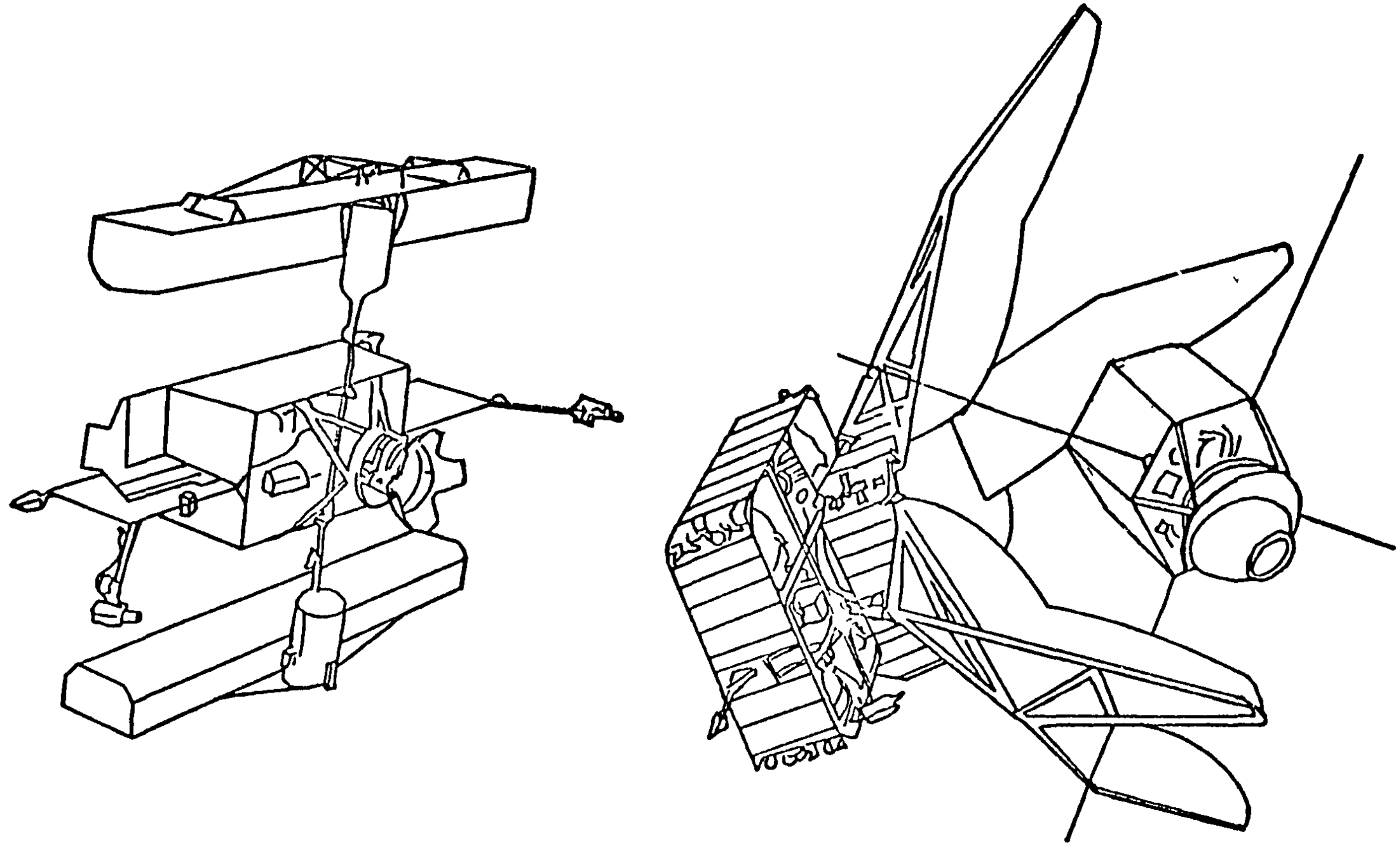


FIGURE 4. ORBIT DEBRIS CONTROL CONCEPT(4)

Summary

In the long term (~15-30 years), there are a variety of potential low-thrust Earth orbital applications that look promising. There is good potential for use of low thrust in any scenario that includes a greatly expanded level of Earth orbital activity, probably including manned space stations. However, the level of increased activity required is unlikely to occur in the 1980s, and from the standpoint of the objectives of the present assessment, it is the 1980s which are of most interest

In the 1980s, the most likely Earth orbital application of a low-thrust propulsion system is delivery of more or less conventional automated spacecraft (relatively small in size) and, possibly, some much larger special purpose payloads (e.g , large antennas) that may evolve after routine Shuttle operations are established. For low-thrust delivery, the destination of interest is geosynchronous orbit. Other Earth orbits have much lower energy requirements and are better suited to chemical propulsion. For these reasons, it was decided that, for the present study, geosynchronous payload delivery would be the primary mission against which low-thrust system capabilities would be evaluated.

LOW-THRUST MISSION PROFILES

For the Ion Drive system, two geosynchronous mission profiles were considered. Both are illustrated in Figure 5. Profile "A" is designed to keep the Ion Drive system out of the radiation belts as much as possible, so as to minimize solar cell damage. A reusable Ion Drive stage is stationed in an "exchange orbit" (~15,000-16,000 km) above the most damaging region of the belts. For a typical mission, the Shuttle would carry an IUS plus a stack of payloads to low orbit (300 km altitude, 28° inclination). The IUS would then deliver the payload stack to the exchange orbit (~16,000 km altitude, 14° inclination), where the payloads would be transferred to the Ion Drive stage. The Ion Drive would transport the payload stack to geosynchronous orbit (35,900 km altitude, 0° inclination), deploy the individual payloads, and return to the exchange orbit to pick up another stack as required. The Ion Drive portion of the delivery (excluding deployment and return) is a spiral trajectory of 60 to 110 days duration

Advantages of Profile "A" are that transit times are relatively low (60-110 days), and radiation damage is minimized. Its principal disadvantage is that it requires the use of two propulsion systems (the IUS and the Ion Drive) to complete the payload transfer from the Shuttle to geosynchronous orbit.

Profile "B" eliminates the need for the IUS. The Ion Drive operates directly from the Shuttle, spirals all the way through the radiation belts and delivers a payload or stack of payloads to geosynchronous orbit. For this profile, the solar arrays must be overdesigned to absorb radiation damage, reusability and stage lifetime are adversely affected, and transit times

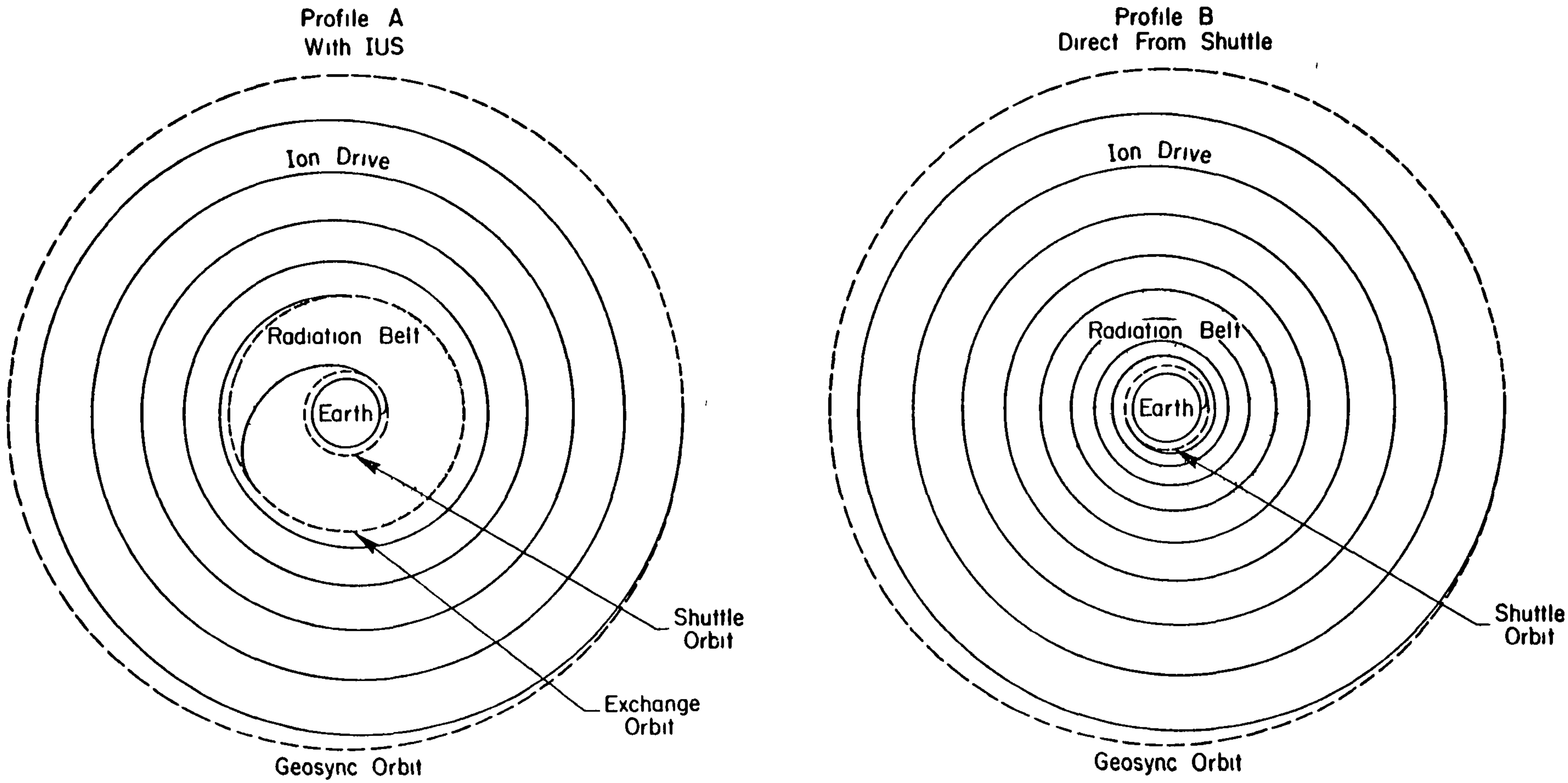


FIGURE 5. ION DRIVE GEOSYNCHRONOUS MISSION PROFILES

are increased to over 200 days. This profile offers potential transportation cost savings compared to Profile "A". However, these savings will be offset by increased costs in the payload programs due to the lengthy transit time and increased exposure of the payload to the radiation belts. Profile "B" is probably best suited for one-way transfers of new, large payloads.

Figure 6 illustrates the Solar Sail mission profile studied. The Solar Sail cannot operate directly from the Shuttle because of its sensitivity to aerodynamic drag--below 1,000 km the aerodynamic drag force on the Sail surface begins to exceed the thrust force of solar light pressure. However, unlike Ion Drive, Sail performance is unaffected by passage through the radiation belts.* Therefore, it would be advantageous to station the Sail in an exchange orbit only slightly above the 1,000 km limit. This minimizes chemical propulsion requirements and the cost of transferring payloads from the Shuttle to the Sail.

The Sail mission profile thus has characteristics similar to both the "A" and "B" profiles. It has a much smaller intermediate chemical propulsion requirement than the Ion Drive "A" profile. However, like the "B" profile, it involves extended payload travel through the radiation belts and lengthy delivery times.

*Performance is unaffected, but lifetime is affected. Radiation exposure gradually reduces the strength of Sail material, rendering it unreliable after several passes through the belts.

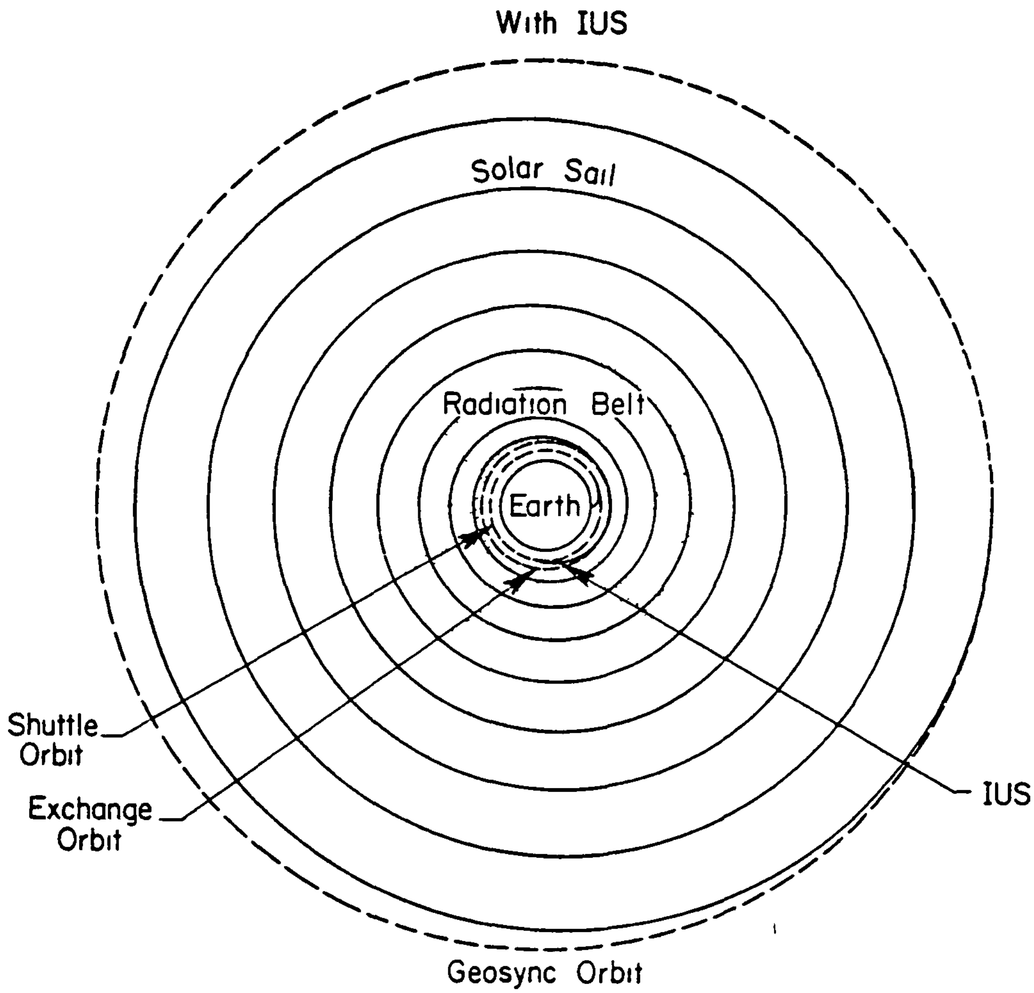


FIGURE 6. SAIL GEOSYNCHRONOUS MISSION PROFILE

PERFORMANCE

Performance of the Ion Drive system and the Solar Sail were analyzed based on the mission profiles just described. Due to significant differences in the thrust vector pointing capabilities of the two systems, different methods of analysis were required.

For the Ion Drive system a set of performance approximation relationships was developed by considering the equations of motion as written in terms of orbital elements. For Mission Profile "A", where radiation damage is minimal, the use of several approximations enabled a closed form solution of the equations of motion (see Appendix A for details). The resulting algebraic expressions determine the flight time required to simultaneously accomplish specified orbit raising and inclination reduction operations. For Profile "B" Ion Drive Missions, where radiation damage is extensive and shadowing times are longer, results from the Profile "A" analyses were incorporated into a computer program that calculates flight time in steps from low orbit, through the belts, and into geosynchronous orbit. This enabled more accurate modeling of the effects of shadowing and reduced power from radiation damage These two performance analysis techniques were very useful because they permitted rapid analysis of a large number of cases, greatly facilitating the Ion drive performance evaluation.

Solar Sail performance analysis is much more difficult Simplifying approximate solutions, such as those applied to Ion Drive, do not currently exist. The inherent constraints on Sail thrust vector pointing complicate the performance analysis problem (the Sail cannot generate a thrust component toward the Sun, and Sail turning rate capability for thrust vector reorientation is extremely limited).

At the time this study began, there was no performance analysis program available that would analyze Earth orbital performance of the Sail. However, there were several programs in existence that analyzed Sail planetary performance. One of these, the "THRUST" program, was modified to handle the Earth orbital case.

THRUST is a numerical integration three-degree-of-freedom trajectory code with open-loop, preprogrammed steering (no optimization). Trajectories generated by the program generally do not represent the shortest possible flight times. Generation of a complete trajectory for low orbit to geosynchronous orbit is a lengthy process, consequently, the Sail performance information which has been generated is very limited

Through more than half the study period, the JPL baseline Solar Sail configuration was the square sail. Most of the performance analysis work was on that configuration, for which two trajectories were generated Neither trajectory modeled turning constraints, and neither was optimized. Nonetheless, based on this information, it was possible to gain some knowledge about the general level of performance of Solar Sails in Earth orbit, including estimates for the current baseline system, the Heliogyro. More work is needed before a full evaluation of Sail performance in Earth orbit can be made.

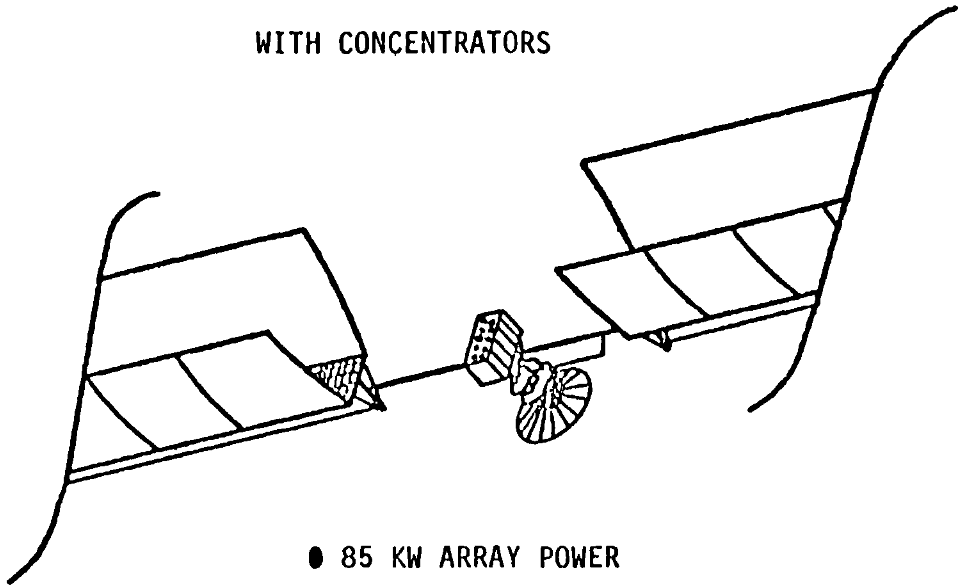
System Models

Two Ion Drive configurations were analyzed, one for each of the two postulated Ion Drive Earth orbital mission profiles Both were derived from the proposed Halley propulsion module. The primary difference between the two configurations is in the solar arrays. Both would use the same number of solar cells as the Halley module, but one would include concentrators (2.1 geometric concentration ratio) while the other would not

The characteristics of both systems[1,5] are defined in Figure 7. The masses quoted in Figure 7 are representative of fully automated and reusable stages, including a rendezvous and docking capability. Both stages would use the same number of thrusters (8 active + 2 reserve = 10) and operate at a maximum power level of 48 kw. However, the stage with concentrators would be capable of generating 86 kw at beginning of life (BOL). The excess power would be used as a cushion to absorb damage in the radiation belts and maintain the capability to operate the thruster system at its rated 48-kw power level as long as possible One trip through the belts would cut array power to the range of 40-50 kw. Subsequent trips would do progressively much less damage. The stage configuration without concentrators would generate 58 kw (BOL) and would be intended for use only outside the more damaging regions of the radiation belts.

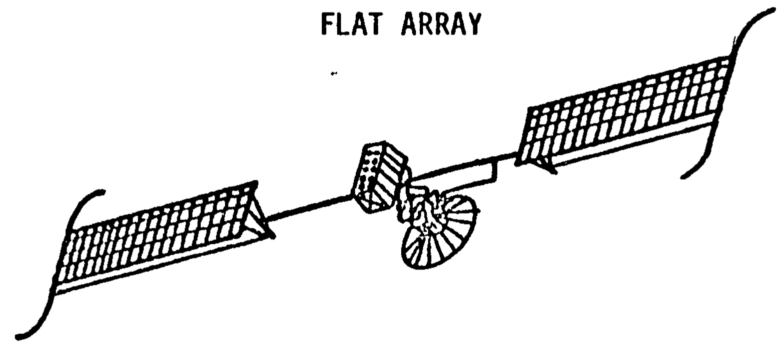
Sail system definition[2,5] is presented in Figure 8 The size of the sail might vary depending on the application. For the present study, Sail size was assumed to be the same as that of the Halley Sail module. As indicated, the conversion of the Sail module to a stage would increase Sail mass by an estimated 500 kg to accommodate guidance, navigation, communications, power, and docking systems

WITH CONCENTRATORS



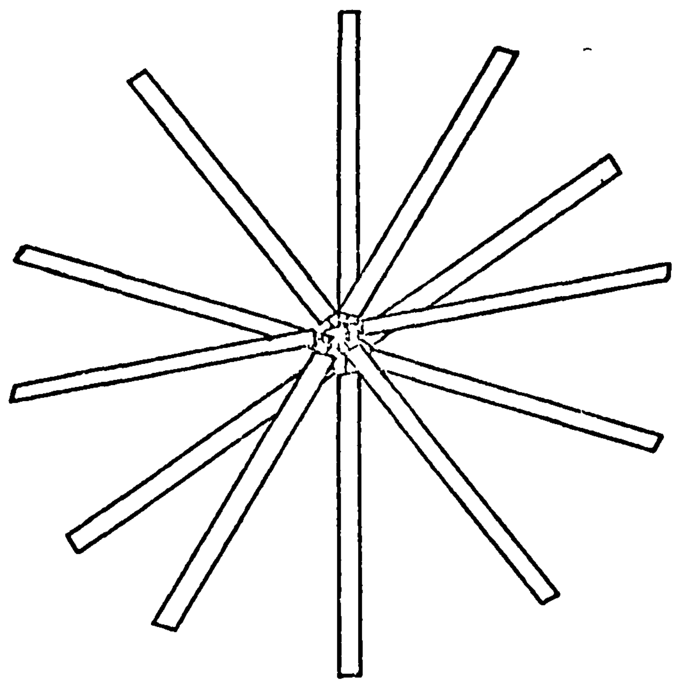
- 85 KW ARRAY POWER
- 10 THRUSTERS
  (8 + 2 RESERVE)
- 48 KW MAX. USED
- 0.73 EFFICIENCY
- 4560 SEC I_{SP}
- 1900 KG DRY MASS

FLAT ARRAY



- 58 KW ARRAY POWER
- 10 THRUSTERS
  (8 + 2 RESERVE)
- 48 KW MAX. USED
- 0 73 EFFICIENCY
- 4560 SEC. I_{SP}
- 1683 KG DRY MASS

FIGURE 7. ION DRIVE EARTH ORBITAL SYSTEMS[1,5]



- SIZE VARIABLE
- HALLEY DERIVATIVE
  - 625,000 M^2 AREA
  - 4000 KG + (500 KG)
  - 1.05 MM/SEC^2 a_0

FIGURE 8. SOLAR SAIL EARTH ORBITAL SYSTEM(2,6)

System Performance

Ion Drive - Mission Profile A

Profile A requires the use of the IUS. The IUS is assumed to deliver a payload stack to an exchange orbit at ~16,000 km* where the payloads are transferred to the Ion Drive system. Thus, the exchange orbit payload delivery capability of the IUS is a major factor in determining the overall performance of the IUS/Ion Drive combination.

Current planning includes four standard multiple-stage IUS configurations. Three of the configurations are derived entirely from different combinations of the two projected IUS solid rocket motors a 10,300-kg "large" motor, and a 2900-kg "small" motor. These configurations are.

(1) Two-stage IUS -- one large motor, one small motor

(2) Twin-stage IUS -- two large motors

(3) Three-stage IUS -- two large motors, one small motor

The fourth configuration is a four-stage vehicle constructed by adding a TE-364 spin stage to the three-stage configuration. It is intended for Pioneer-class spin-stabilized planetary spacecraft. Because the final stage is spun, this four-stage configuration is not suitable for the IUS/Ion Drive mission application.

The performance capabilities of the remaining three configurations were evaluated. Each of the IUS motors provides a fixed impulse** Configuration performance is dependent on how well the velocity increments resulting from these impulses match the two velocity changes required to transfer from the Shuttle orbit to the IUS/Ion Drive exchange orbit.

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* 15,000 to 16,000 km is the lower end of the operating regime for Ion Drive, if excessive array damge is to be avoided.

** The motors do not have a stop/restart capability, once ignited, a motor burns to propellant depletion. Some reduction in motor impulse can be obtained by offloading propellant

For the twin-stage IUS the match is poor, thus, the twin-stage configuration is not practical in this application. The two-stage match is much better It can deliver a 3,800-kg payload to a 16,000-km orbit at 14° inclination The three-stage IUS cannot improve on the payload mass, it can only further reduce the inclination. This is no practical advantage because the Ion Drive can easily accomplish inclination reduction. Given that the payload delivery capabilities of the two-stage and three-stage IUS's are the same, the two-stage IUS is the better choice because of its shorter length requirement in the Shuttle cargo bay.

Combinations of the IUS motor, other than the standard configurations, can be envisioned and would be potentially useful One of these is a modified four-stage configuration consisting of two large IUS motors and two small IUS motors. To satisfy Shuttle payload constraints, all four stages must be offloaded. This configuration could deliver 5,600 kg to a 16,000-km exchange orbit at 14° inclination. As a result of its significantly greater payload delivery, this modified four-stage IUS was carried in the analysis as an alternative to the two-stage IUS. However, the increased length of the four-stage configuration is a disadvantage from the standpoint of Shuttle loading.

Performance of the IUS/Ion Drive system is shown in Figures 9 and 10. Figure 9 presents resulting using the two-stage IUS and Figure 10 presents those for the modified four-stage IUS. In both cases, the 58-kw flat array Ion Drive is assumed, along with the assumption that the Ion Drive system is refueled with Mercury propellant for each trip. The required propellant mass (~350 kg in the case where the two-stage IUS is used) was deducted from the IUS capability to determine the maximum payload capability as expressed by the "IUS Limit" shown on the figures. Ion Drive flight time is plotted as a function of payload. Both payload delivery time and the Ion Drive round-trip (payload delivery and stage return) time are shown. All trips originate and terminate at the IUS/Ion Drive exchange orbit

As shown, for the two-stage IUS case, delivery of the 3000+ kg payloads requires 60 to 80 days. Round-trip time is 80 to 110 days. For the four-stage IUS case, delivery of its ~5000-kg payloads requires 85 to 110 days, with round-trip time being 105 to 135 days.

When the cost of using the Ion Drive for geosynchronous missions is compared to that of competing chemical systems, the lifetime of each Ion Drive stage is a key factor. For a given round-trip time, the lifetime determines the number of missions that can be performed by each stage and, hence, the number of missions over which the cost of the stage may be amortized. Ion Drive lifetime is limited by ion thruster operating life, which has an estimated maximum of 20,000 hours, or 833 days. As shown in Figure 9, for the two-stage IUS case 8 to 10 trips are possible, depending on the mass of the payloads. In Figure 10, 6 to 8 trips are shown as possible for the four-stage IUS case. Since the payload ranges are small (a few hundred kilograms), from

2 STAGE IUS/ION DRIVE GEOSYNC DELIVERY PERFORMANCE (PROFILE A)

- IUS DELIVERS PAYLOAD ONLY TO EXCHANGE ORBIT (~16,000 KM/14°)
- ION DRIVE TAKES PAYLOAD FROM EXCHANGE ORBIT TO GEOSYNC
- 58 KW ARRAY POWER (BOL)
- 48 KW MAX PROPULSION INPUT POWER
- ~350 KG PROPELLANT PER TRIP

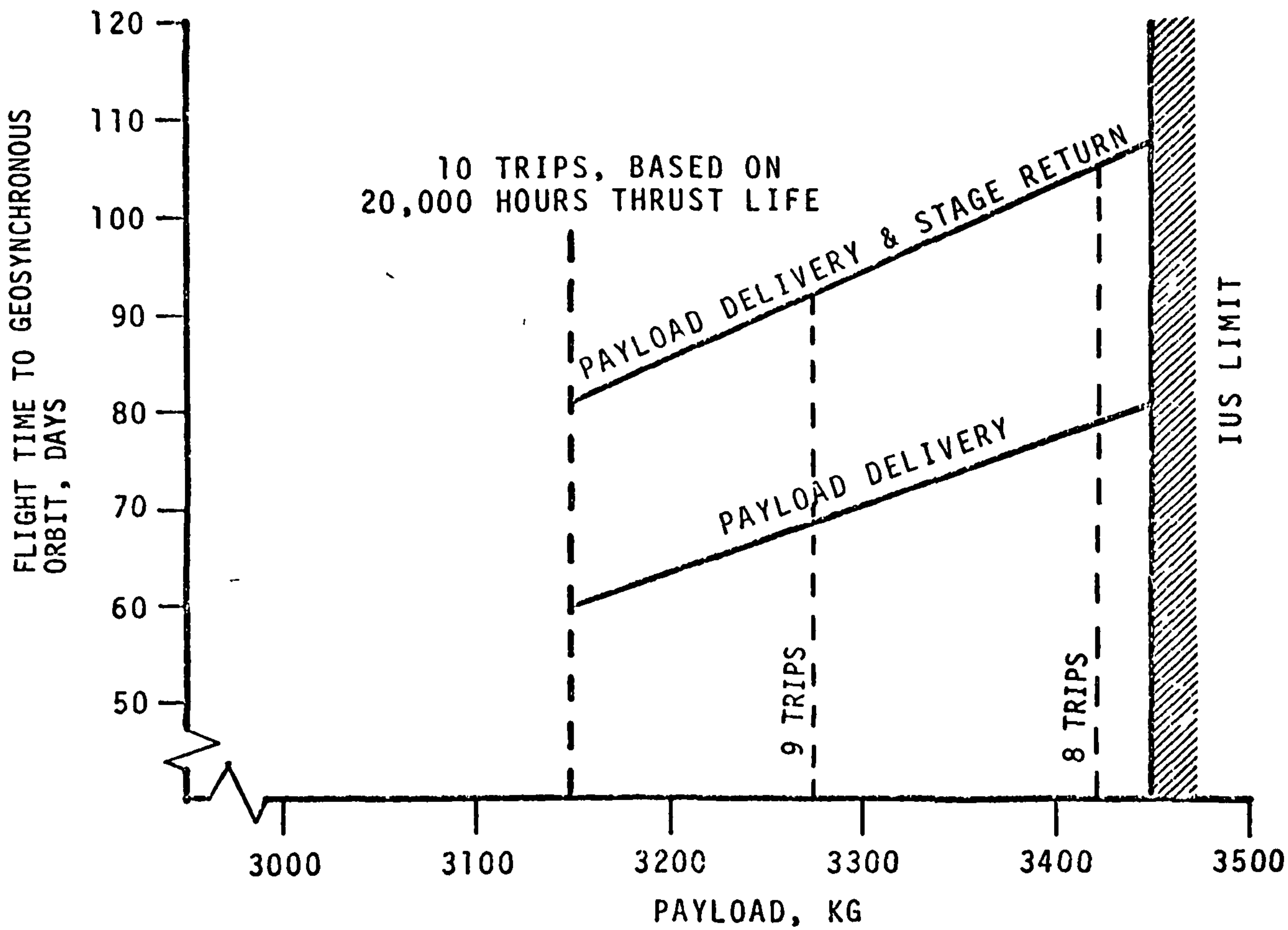

FIGURE 9. ION DRIVE PERFORMANCE (58 KW WITH 2-STAGE IUS)

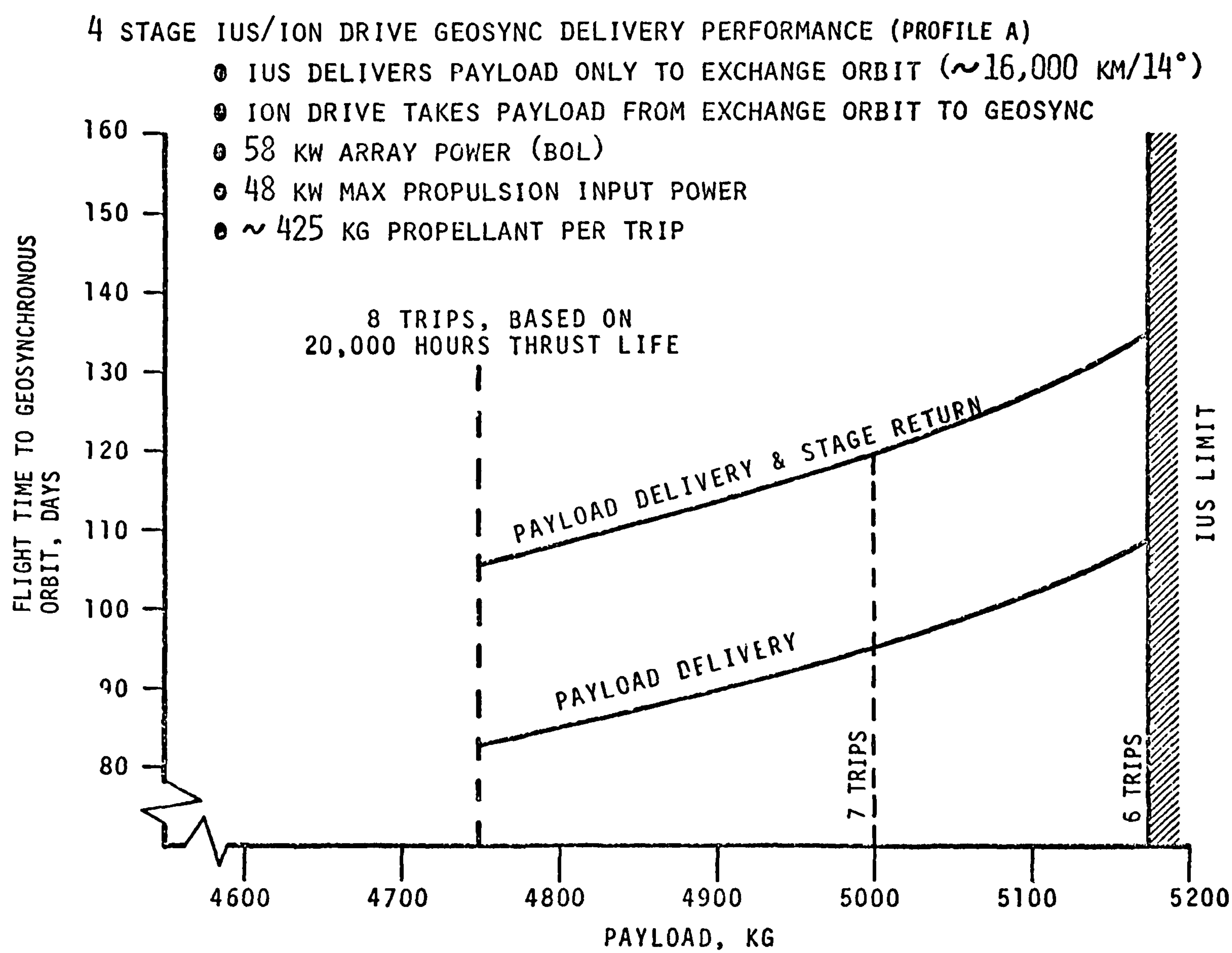

FIGURE 10. ION DRIVE PERFORMANCE (58 KW WITH 4-STAGE IUS)

an economic standpoint it would be better in both cases to operate at the lower payload values and maximize the number of trips per stage.

Ion Drive - Mission Profile B

Profile B assumes the Ion Drive operates directly from the Shuttle. The IUS is not required. Array power is increased to 85 kw (BOL) through the addition of concentrators.

Geosynchronous mission performance is shown in Figure 11, where flight time is plotted as a function of payload mass for a wide range of payloads. The flight time shown is the delivery time only Multiple trips are probably not practical for the currently defined system due to the combined effects of high flight times and solar array degradation. At best, only two or three deliveries of small, 1000-2000 kg, payloads could be made before the 20,000-hour thruster lifetime would be exceeded If the Ion Drive is operated directly from the Shuttle, it is better suited to missions where it is used as a dedicated propulsion system for delivery only of large payloads

Solar Sail

As noted earlier, two Solar Sail Earth orbital trajectories have been generated. Both assumed a Square Sail, although Heliogyro mass and area parameters were used. Because of computer program limitations, the trajectory simulations did not include the rather severe turning constraints that exist even with the Square Sail, and neither of the trajectories was optimized. These program limitations have opposite effects on the performance calculation and it is not clear what the net effects would be

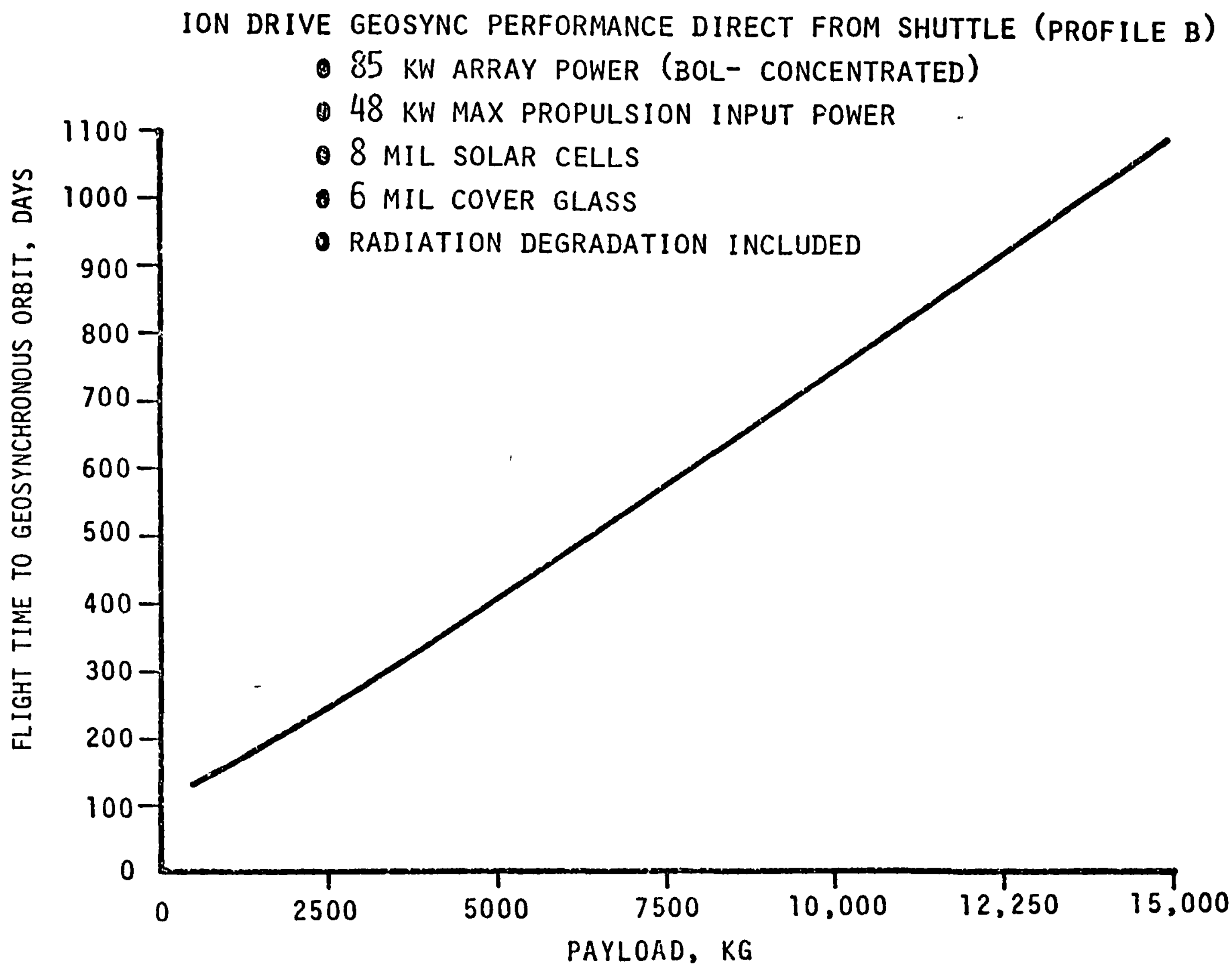

FIGURE 11. ION DRIVE PERFORMANCE (85 KW DIRECT FROM SHUTTLE)

The Sail ascent to geosynchronous orbit was assumed to start from 1,500 km to avoid the aerodynamic drag problem present with lower orbits. An additional propulsion system would be needed to transfer the Sail and/or payloads from the Shuttle to 1,500 km. The energy requirement is small but two velocity impulses are required. If non-restartable solids are to be used, then two motors will be required. None of the standard IUS configurations are appropriate, because they provide much more energy than is needed. A modified IUS, using twin small IUS motors, could do the job, although it, too, delivers more performance than is likely to be needed (20,000 kg to 1,500 km).

Characteristics of the two Sail trajectories are summarized in Table 1 below. In generating both trajectories, synchronous orbit was attained long before the necessary 28° inclination reduction was completed. Only a 10° reduction was accomplished. The remaining 18° was taken out by assuming continued operation of the Sail after synchronous orbit had been attained, until 0° inclination was achieved.

TABLE 1. SQUARE SAIL[a] GEOSYNCHRONOUS MISSION PERFORMANCE

	First Trajectory	Second Trajectory
Payload	3500 kg	15,500 kg
Time to Synchronous Altitude at 18° Inclination	450 days	750 days
Time to 0°	85 days	225 days
Total Time to Geosynchronous Orbit	535 days	975 days

(a) Using Heliogyro mass and area parameters, excluding turning constraints, and not optimizing performance.

Although the data in Table 1 are for a Square Sail, they can be used to obtain a projection or first estimate of the level of performance of a Heliogyro Sail.

To a first approximation, the Heliogyro may be treated as a Square Sail with no turning constraint, but with reduced effective Sail area. The Heliogyro as a whole has a much slower turning rate* than the Square Sail. However, the individual blades can be cylically pitched to provide force vector pointing control not available with the Square Sail. This control is achieved at the expense of a reduction in force vector magnitude, which may be interpreted as a reduction in effective Sail area.

If a fairly liberal view of the capabilities of cyclic pitch is adopted, and some design charges are incorporated, a reduction in effective Sail area to 75 percent of its original value can be estimated. Based on this estimate and the Square Sail trajectory data, a Heliogyro performance projection was generated The results are shown in Figure 12.

In that figure, payload is plotted as a function of flight time The lower line represents the Square Sail estimates as determined by the two trajectories summarized in Table 1. Again, these data are based on a Square Sail with the same mass and area as the Heliogyro. Thus, $M_S = M_H$ and $A_S = A_H$, where M and A are the mass and area of the Square Sail (subscript S) and the Heliogyro (subscript H), respectively.

*Maximum Heliogyro turning rate is 0.6 degrees per hour. This makes the Heliogyro virtually immobile over one orbit around Earth (a few hours duration). However, the 0.6 deg/hr rate is more than adequate to track the Sun for several months--or years, if necessary

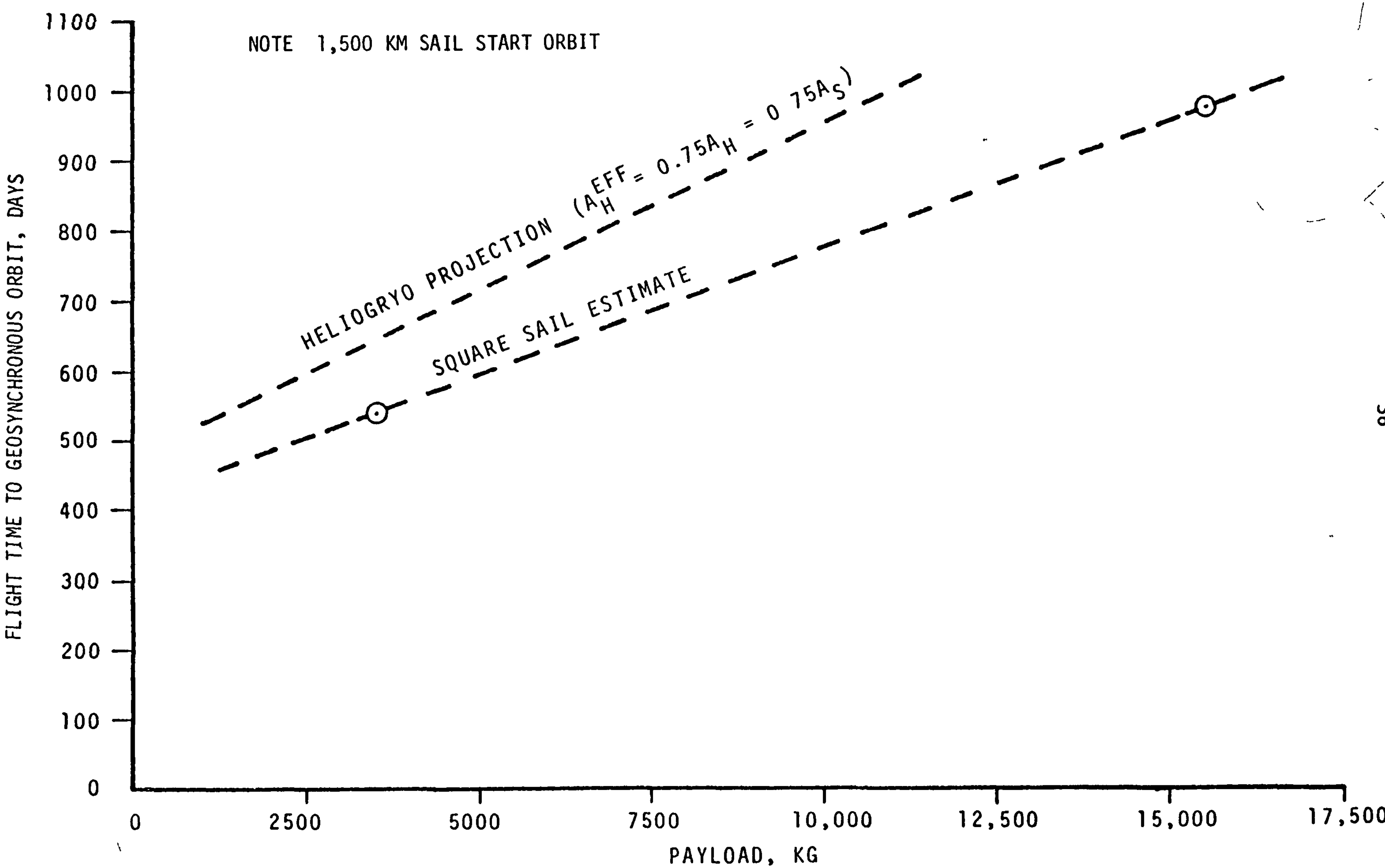

FIGURE 12. SAIL GEOSYNCHRONOUS DELIVERY PERFORMANCE ESTIMATES

The upper line in Figure 12 is the Heliogyro projection, assuming the effective Heliogyro Sail area (A_H^{eff}) is 0.75 times the Square Sail area (A_S). The projection was determined from the Square Sail data by calculating the payload reduction required to keep flight time unchanged when Sail area is reduced. The calculation is performed by holding the ratio of Sail area to total mass (Sail mass plus payload) constant when the Sail area is reduced. This keeps the force-to-mass ratio (acceleration) unchanged so that the trajectories remain the same, and flight time is not changed. If M_{PLS} is the Square Sail payload and M_{PLH} is the Heliogyro payload, then the calculation proceeds as follows:

$$\frac{A_H^{eff}}{M_H + M_{PLH}} = \frac{A_S}{M_S + M_{PLS}} \;;$$

and $$A_H^{eff} = 0.75\, A_S,\; M_H + M_S \;,$$

therefore $$\frac{0.75\, A_S}{M_S + M_{PLH}} = \frac{A_S}{M_S + M_{PLS}} \;,$$

solving for M_{PLH},

$$\begin{aligned} M_{PLH} &= 0.75\,(M_S + M_{PLS}) - M_S \\ &= 0.75\, M_{PLS} - 0.25\, M_S \end{aligned}$$

The results indicate that Sail flight times are high. Over a year is required to reach geosynchronous orbit, even for small payloads. Optimization of trajectories could substantially reduce flight times. On the other hand, the Square Sail estimate ignores turning constraints and the 0.75 area reduction logic used to obtain the Heliogyro projection is probably optimistic. A more complete analysis is needed before more definite conclusions can be reached.

Ion Drive/Solar Sail Comparison

Ion Drive and Solar Sail geosynchronous delivery performances are compared in Figure 13. The Ion Drive curve represents Mission Profile B, where the Ion Drive operates directly from the Shuttle. The Sail performance assumes a 1500-km Sail start orbit and, therefore, involves the use of another propulsion system operating between the Shuttle and the Sail start orbit. Ion Drive Profile B was selected for the comparison because its low-thrust segment more closely corresponds to the Sail profile than does Ion Drive Profile A.

As shown, the analyses indicate that Ion Drive flight times would be less than those of the Heliogyro (significantly less at low-to-moderate payload values). Optimization should improve the Heliogyro performance estimate, but this could be offset by operational constraints not fully included in the current performance estimates.

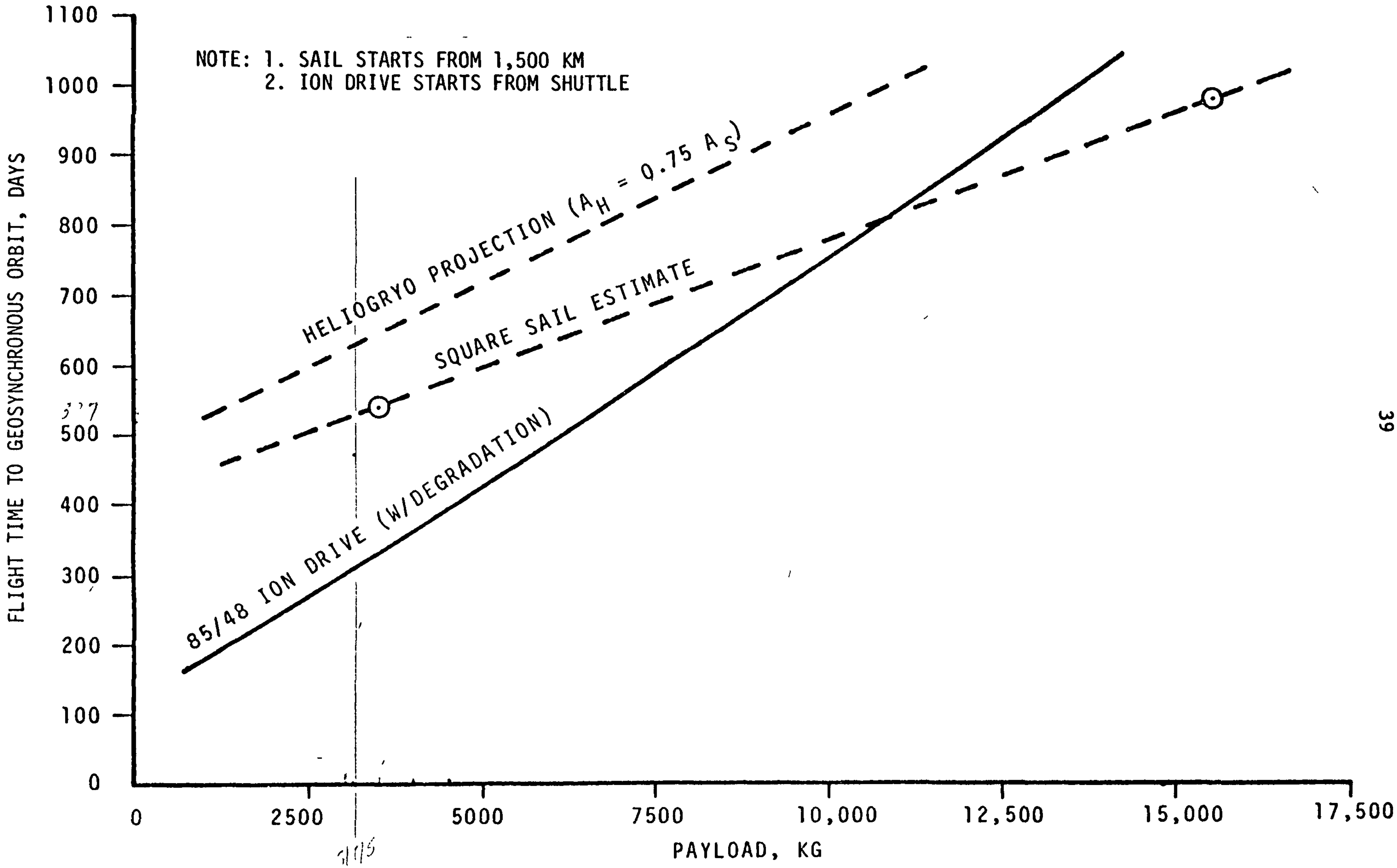

FIGURE 13. ION DRIVE/SOLAR SAIL PERFORMANCE COMPARISON

In the geosynchronous Earth orbital application, the Sail appears to fare less well in comparison to Ion Drive than it does on the Halley mission. The reason for this is primarily that the force vector pointing constraints inherent in the Sail design are much more restrictive in Earth orbit than they are in heliocentric space, where the Sail is always moving around the Sun and its direction of motion with respect to the Sun line changes slowly. In Earth orbit, the Sail direction of motion with respect to the Sun line assumes a wider range of values, and changes occur much more rapidly. In Earth orbit, Sail motion may be directly towards the Sun. When it is, the Sail can generate no force contributing to orbit raising. The Sail is at a significant disadvantage in Earth orbit, and this fact is reflected in the performance estimates

Performance Sensitivity

The sensitivity of Ion Drive and Solar Sail performance data to changes in system characteristics was investigated. Potential performance degradation due to operational flight hardware not meeting design specifications was analyzed, as well as the performance growth potential afforded by selected design/system improvements

For the Ion Drive system, three possible sources of performance degradation are loss of thruster efficiency, weight growth, and loss of thruster lifetime. For the IUS/Ion Drive operational mode (Mission Profile A), loss of thruster efficiency and/or system weight growth increases flight time, resulting in a decreased number of trips possible

with each stage. Loss of thruster lifetime produces the same result directly. The primary impact in all cases is economic a reduced number of trips per stage, resulting in increased cost per trip.

Table 2 shows the decrease in number of trips per stage caused by thruster efficiency loss and weight growth for Ion Drive Mission Profile A. Data are given for 10 and 20 percent decreases in thruster efficiency and 10 and 20 percent weight increases. Results applicable to the Ion Drive mission as defined using either a two-stage or modified four-stage IUS are included. The undegraded performance (baseline thruster efficiency and weight) is shown at the bottom of the table for reference

Summarizing the results shown in Table 2, it appears that in all but one case each 10 percent degradation of a parameter results in the loss of one trip. The exception is the first 10 percent weight growth for the two-stage IUS case, where two trips are lost The relationship between thruster lifetime and number of trips per stage is shown in Figure 14 for Ion Drive Mission Profile A. The currently estimated feasible lifetime limit of an electric thruster system is 20,000 hours. This defines the baseline for number of trips possible at 10 for the two-stage IUS and 8 for the four-stage IUS. If lifetime were cut to the Halley mission value of 15,000 hours, then those numbers would drop to 7 and 6, respectively. Increases in lifetime would bring corresponding increases in number of trips possible

TABLE 2. ION DRIVE PERFORMANCE SENSITIVITY TO THRUSTER EFFICIENCY, STAGE WEIGHT

Degradation Source	Percentage Degradation	IUS Configuration	Trips at 20,000 hr Thrust Life
Thruster	10	2-Stage	9
Efficiency	20	2-Stage	8
Loss	10	4-Stage	7
	20	4-Stage	6
Weight	10	2-Stage	8
Growth	20	2-Stage	7
	10	4-Stage	7
	20	4-Stage	6
None	--	2-Stage	10
(Undegraded)	--	4-Stage	8

NOTE. Mission Profile A, 58/48 Ion Drive operating above heart of radiation belts.

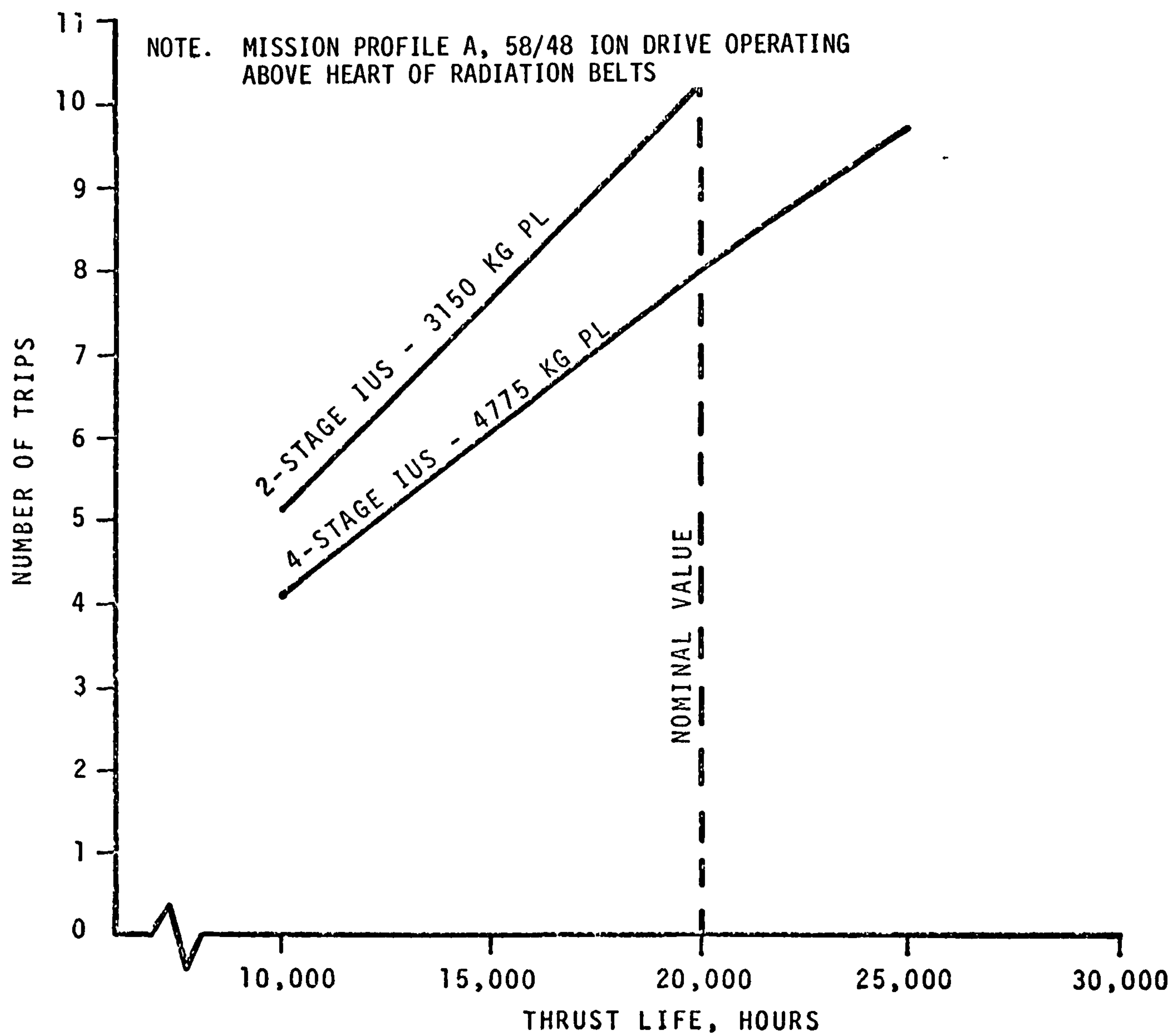

FIGURE 14 ION DRIVE PERFORMANCE SENSITIVITY TO THRUSTER LIFE

For Ion Drive Mission Profile B (direct from Shuttle) and for the Solar Sail, performance sensitivity was defined only in terms of its effect on flight time for a single mission. Results, shown in Table 3, indicate a 10 to 30 percent increase in flight time can be expected for 10 to 20 percent losses in Ion Drive thruster efficiency. The impact due to weight growth is much smaller, 11 percent or less flight time increase for up to 20 percent Ion Drive weight growth.

Thruster efficiency has no meaning for the Sail, but weight growth does. As with Ion Drive, the impact of weight growth is relatively small Flight time increases 8 percent, or less, for up to 20 percent weight growth.

By the mid-1980s, a number of low-thrust system improvements that would upgrade performance might be possible. In the case of the Sail, it may be possible to produce and use significantly thinner Sail film. The Halley design calls for 0 1 mil (2.5 micron) film. Films 1 micron and thinner are thought by Sail designers to be possible and would significantly reduce Sail weight. For Ion Drive, the most promising area of improvement (for Earth orbital missions) would probably be the incorporation of solar cells that do not degrade significantly when exposed to the radiation belts. Solar cells have already been developed that approximate this behaviour They are made from gallium arsenide (GaAs) rather than the silicon now used in conventional cells. Unfortunately, at present, GaAs cells are two to three times more expensive to produce than are the silicon cells.

TABLE 3. ION DRIVE/SAIL PERFORMANCE SENSITIVITY TO THRUSTER EFFICIENCY, STAGE WEIGHT

Degradation Source	Percentage Degradation	Vehicle	Percentage Flight-Time Increase
Thruster Efficiency Loss	10	Ion Drive	10 - 12
	20	Ion Drive	25 - 30
Weight Loss	10	Ion Drive	~0 - 5
	20	Ion Drive	~0 - 11
	10	Heliogyro	3 - 5
	20	Heliogyro	5 - 8

NOTE: Data are for the 85/48 Ion Drive (operating direct from Shuttle) and Heliogyro configurations

The performance increases generated by the above defined improvements are shown in Figure 15. For the Sail, reducing film thickness from 0 1 mil to 1.0 micron cuts flight time 10 to 20 percent (compare curves 1 and 2 for the Heliogyro, and curves 3 and 4 for the Square Sail).

For the Ion Drive, the potential performance gain is observed by comparing curves 5 and 6. Curve 5 represents the baseline silicon cell system with an 85-kw concentrator array but only a 48-kw thrust system. As noted earlier, the power surplus is used to absorb radiation damage and keep the thruster system operating at full power for as long as possible Curve 6 assumes that gallium arsenide cells replace the silicon cells and the thruster system grows to match array power at 85 kw. This produces a dramatic increase in performance (approximately 50 percent reduction in flight time).

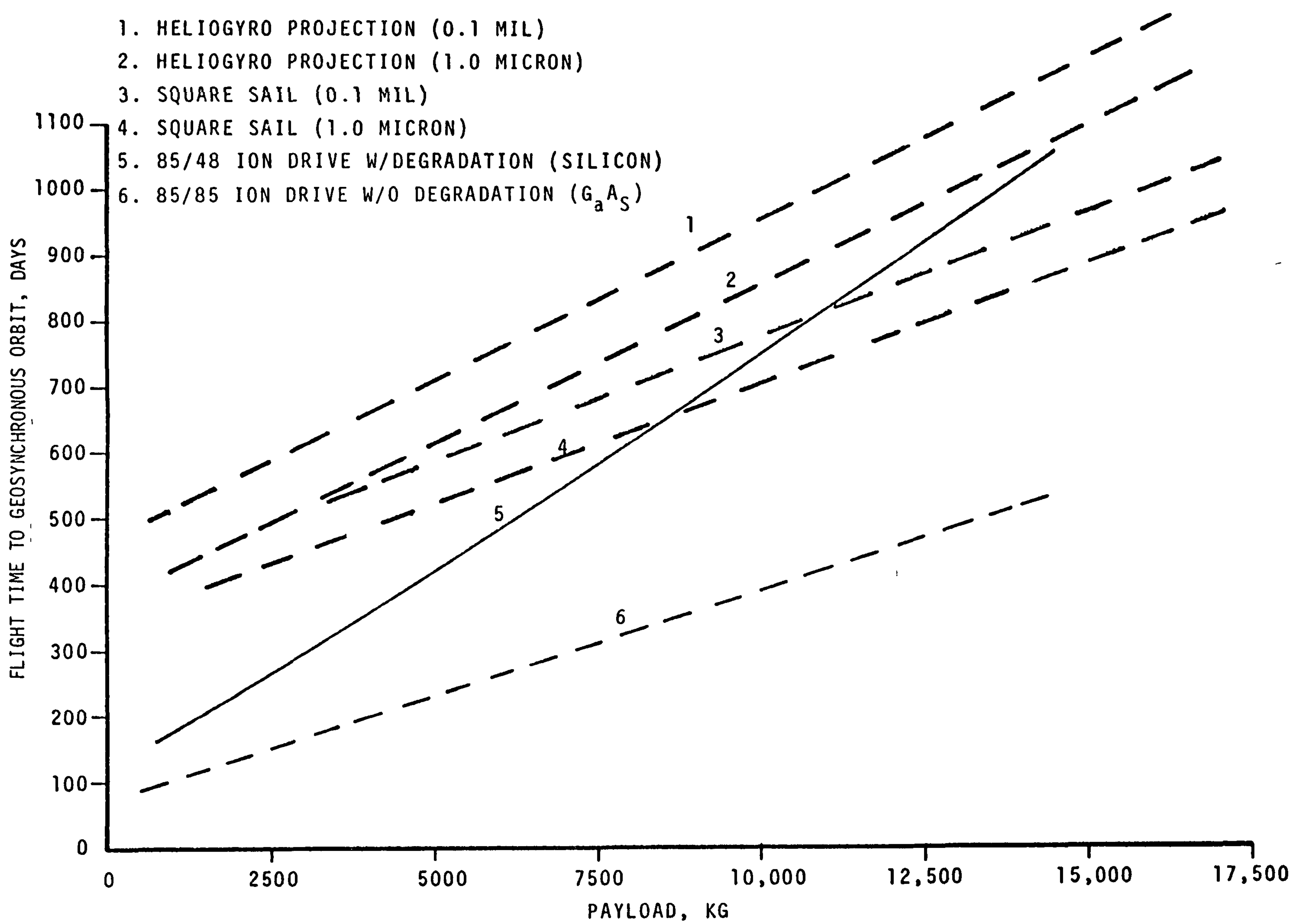

FIGURE 15 PERFORMANCE GROWTH POSSIBILITIES

COMPARISON OF LOW-THRUST AND CHEMICAL PROPULSION

As noted earlier, the most likely Earth orbital application of low-thrust propulsion in the 1980s is payload delivery to geosynchronous orbit. The extent to which low-thrust propulsion might be used in that role will depend on how well it competes with chemical systems such as the IUS and Spinning Solid Upper Stage (SSUS). In most cases, the question becomes one of cost rather than performance. In this section, the cost of transporting payloads to geosynchronous orbit with low-thrust propulsion is compared to that for chemical propulsion.

To perform the desired comparison, definitions of the performance and costs of competing systems must be established Also, a basis for comparison must be established--in this case, an appropriate set of geosynchronous delivery missions to be performed.

Chemical Systems and Performance

The characteristics of expected Shuttle upper stages of interest are summarized in Table 4. Those which are regarded as competitors to low thrust for Earth orbital missions are the two-stage IUS, the three-stage IUS, the Atlas-class SSUS (SSUS-A), and the Delta-class SSUS (SSUS-D) The twin-small-stage IUS is included not as a competitor to low-thrust systems, but as a possible supporting system for the Solar Sail. Likewise the modified four-stage IUS (along with the two-stage IUS), is a possible supporting system for Ion Drive Mission Profile A The twin-stage IUS could support either the Sail or Ion Drive on planetary missions.

TABLE 4. STS CHEMICAL UPPER STAGES

	Support Equipment (kg)	Mass (kg)	Length (m)	Diameter (m)	Geosynchronous Payload (kw)
IUS					
Twin Small	1660	6874	4.0	2.3	--
2-Stage	1660	14419	4 5	2 3	2268
Twin-Stage	1835	21758	5 4	2.3	--
3-Stage	1835	25244	7 0	2 3	3266
4-Stage Modified (2 large, 2 small)	1835	21630 (a)	9.0	2.3	--
SSUS					
A	1792 (b)	3545	2.3	1.3	2041 (c)
D	1270 (b)	1760	1.9	1.2	1111 (c)

(a) All motors offloaded
(b) Estimated values
(c) Payload to geosynchronous transfer only

As shown, the two-stage IUS can deliver a 2268-kg payload into geosynchronous orbit. The three-stage IUS can deliver 3266 kg. In both cases, the delivery is in a three-axis stabilized mode. The SSUS stages are spin-stabilized and only do the transfer portion of the geosynchronous mission. The payloads quoted for SSUS-A (2041 kg) and SSUS-D (1111 kg) must include an apogee motor to circularize and complete the necessary plane change for geosynchronous orbit. Thus, for SSUS-A and SSUS-D, the useful payload available in the final orbit is about half the quoted value

All systems being considered would rely on the Shuttle for delivery to low Earth orbit. Thus, the Shuttle capability limits of 39,500 kg (65,000 lb) cargo mass, 18.3 m (60 ft) cargo length, and 4.57 m (15 ft) cargo diameter, apply Shuttle cargo loading, as determined by the sum total of payload and propulsion system dimensions and masses, is an overriding factor in total transportation costs. For the chemical propulsion systems being considered, the masses and dimension required to determine Shuttle loading are included in Table 4.

System Costs

Estimates of the costs of transportation system elements are summarized in Table 5 These data were derived from a variety of sources. The Shuttle cost of $18.5M is a NASA estimate.[7] The costs of the three standard IUS configurations are based on informal preliminary estimates by Aerospace Corporation. Costs for the modified configuration (twin small stage, four stage) were estimated based on the standard system

costs and the extent of the modifications required. Use of these configurations would bring about a small developmental or non-recurring (NR) charge as indicated to accomplish the necessary modifications. The SSUS costs are based on informal preliminary estimates of commercial user charges as provided by contractors who may produce the stages.

TABLE 5. COST DATA (1977 DOLLARS)*

Item	Cost, $M
Dedicated Shuttle	18.5
IUS - Twin Small Stage ($2M NR)	3.5
- Two Stage	4.0
- Twin Large Stage	5 0
- Three Stage	6.0
- Four Stage ($3M NR)	6.0
SSUS D	2.0
SSUS A	2.5
Ion Drive Modifications	27.0
Sail Modifications	28-30
Ion Drive Unit (Stage)	26.8
Sail Unit (Stage)	**

* Source: Boeing 1975 data, IUS/SSUS studies, ELV operations
**Data not available, $27M assumed

Ion Drive costs were estimated by updating Boeing's 1975 detailed costing of a 25-kw SEPS[3] to include the effects of inflation and the higher power level of the Ion Drive. The same is true of the Ion Drive unit cost which assumes an ongoing production of about two stages per year.

No detailed estimate of the cost required to modify the Sail for Earth orbital operations has ever been made. That cost was estimated to be comparable to that of the Ion Drive and, perhaps, slightly higher, because of the need to add a power system to the Sail module, and because rendezvous and docking is a more difficult problem to solve for the Sail than for Ion Drive. The unit cost of an Earth orbital Sail is even more difficult to estimate than the configuration modification cost. The modifications primarily consist of the addition of conventional systems (the costs of which are reasonably well known) to the Sail Halley propulsion module. The Sail unit cost is dominated by the cost of the Sail module itself, about which little is known other than the estimate for the first unit for the Halley mission. For purposes of comparison to chemical propulsion, the Sail unit cost for a ~2/year use rate was assumed to be comparable to that for Ion Drive (~$27M).

Mission Model

If the Halley mission is conducted, then a low-thrust propulsion module would become available in the early 1980s. By the mid-1980s, a complete stage could evolve that would begin to compete for geosynchronous missions through the last half of the decade.

A 1985-1990 geosynchronous mission model was constructed based upon the August 1977 revised Outside User's Payload Model.[8] This model includes projected Shuttle launched missions that would be conducted for private corporations, foreign governments, international organizations, and U.S. Government agencies other than NASA and the Department of Defense (DoD).

NASA no longer conducts many geosynchronous mission programs. These have largely become the province of the user organizations such as INTELSAT and the National Oceanographic and Atmospheric Administration (NOAA). NASA's role in this arena is that of a development organization exploring totally new concepts. For the 1980s, this may mean a limited number of flight experiments relating primarily to space solar power stations (SSPSs) and large communications antennas. These spacecraft would differ significantly from the outside user spacecraft that will dominate geosynchronous mission traffic. The extent to which NASA will pursue these options is not clear at present and they have not been included in the mission model. However, should they become a reality, they may represent unique opportunities for the application of low-thrust propulsion.

DoD missions are not included in the model because they are classified. DoD traffic to geosynchronous orbit is expected to be two or three missions per year, at most. Increasingly, the government is expressing a preference that the DoD lease communications services on commercial spacecraft which are already included in the mission model. However, it should be noted that the DoD is also expressing interest in the large communications antennas mentioned earlier as a new mission concept. Use of the large antennas could greatly improve mobile troop communications.

Another class of geosynchronous missions that have not been included in the mission model includes all those planned for launch on the European Ariane or Japanese "N" launch vehicles. These missions are not expected to become candidates for low-thrust propulsion unless these two launch vehicle programs are cancelled.

Details of the Outside User Geosynchronous Payload Model are presented in Appendix B. All of the spacecraft in the model may be placed in one of four classifications representing two different levels of delivery propulsion system performance requirements (labeled SSUS-A or SSUS-D for convenience), and two different spacecraft design philosophies [dual compatible with the Shuttle and Expendable Launch Vehicles (ELVs), or optimized for the Shuttle]. The dual compatible spacecraft are longer and smaller in diameter than their Shuttle optimized counterparts so that they could fit within existing ELV shrouds and be launched on ELVs, if a Shuttle launch could not be obtained Shuttle optimized spacecraft are short with large diameters to make maximum use of Shuttle cargo space and minimize Shuttle cargo charges.

The designation of performance requirement as SSUS-A or SSUS-D level is not meant to imply that the SSUSs are the preferred propulsion system for all spacecraft in the model. Clearly, the IUS or some other system may be preferred by the spacecraft designers. The SSUS-A and SSUS-D performance level labels are merely intended as useful identifiers of two basic classes of automated geosynchronous spacecraft (Atlas class and Delta class) that have evolved over the years, and that are continuing to dominate spacecraft plans and designs for the early 1980s.

Table 6 summarizes the mission model, grouping all spacecraft in the four classifications defined above and identified as shown in the legend. Launch rates average ten per year over the 6-year 1985-1990 period. In the early years, all spacecraft are dual compatible. In the later years, more than half are Shuttle optimized.

Cost Comparison for Earth Orbital Operations

Low-thrust and chemical propulsion options for the 1985-1990 mission model have been compared. Payload transportation requirements and associated costs were investigated for the baseline model as defined previously and in Appendix B. Requirements and costs also were investigated for a variation from the baseline that assumes all payloads in the model are Shuttle optimized to improve Shuttle payload packaging.

The IUS/Ion Drive combination system (Mission Profile A) was selected as the primary low-thrust option to compare to all-chemical systems. This selection was made for several reasons. With the Ion Drive primarily operating above the radiation belts, supported by the IUS, 8 to 10 uses per Ion Drive stage can be achieved, and payload delivery times are only 2 to 4 months. Using the Ion Drive direct from the Shuttle, or the Solar Sail from a 1500-km start orbit, flight times increase to a year or more and reuse capabilities diminish considerably. A maximum of two to three uses per stage could be obtained due to damage caused to both systems by the radiation belts.

TABLE 6. 1985-1990 PAYLOAD MODEL

SPACECRAFT	LAUNCH SCHEDULE						MASS, kg		LENGTH	DIAMETER
CLASSIFICATION	85	86	87	88	89	90	W AKM	W/O AKM	m	m
SSUS D	5	6	6	3	4	4	570 - 950	285 - 475	3.4-3.8	1.9-2.2
SSUS D'	1	0	0	2	4	5	1090	545	0.9	3.7
SSUS A	4	2	2	3	-	2	1860 - 2040	930 - 1020	5.6-7.2	2.6-2.8
SSUS A'	-	-	2	1	2	2	2040	1020	1.5	4.3
TOTAL LAUNCHES	10	8	10	9	10	13				

$\Sigma = 60$

LEGEND

SSUS D ----SSUS D PERFORMANCE LEVEL, DUAL COMPATIBLE (SHUTTLE AND ELV)

SSUS D'----SSUS D PERFORMANCE LEVEL, SHUTTLE OPTIMIZED

SSUS A ----SSUS A PERFORMANCE LEVEL, DUAL COMPATIBLE (SHUTTLE AND ELV)

SSUS A'----SSUS A PERFORMANCE LEVEL, SHUTTLE OPTIMIZED

If the Sail start orbit were raised to the 15,000-16,000 km level, then the Sail might produce results comparable to the Ion Drive system operating from that altitude. Sail unit costs presently are not well defined and have been assumed to be comparable to Ion Drive costs, pending the availability of additional information. Thus, results obtained comparing the Ion Drive to chemical propulsion are assumed to apply also to the Sail.

Determination of the transportation requirements and costs associated with the 1985-1990 mission model can be reduced to analysis of the 1985 payload set. Results for that year illustrate the general tradeoffs and comparisons for the entire baseline mission model.

The 1985 payloads are defined in Table 7. Efficient use of the Ion Drive system depends on the use of multiple payload stacks to minimize the number of flights required. Two ways of combining the 1985 payloads into stacks was considered. One was to combine all ten payloads into two stacks, identified as flights A and B, below

Flight	Payload Number (as in Table 7)		Mass/Length
A:	1+2+5+6+9	=	3115 kg/16.9 m
B:	4+8+3+7+10	=	3215 kg/15.1 m.

The other option was to go to three stacks, identified as flights C, D, and E.

Flight	Payload Number		Mass/Length
C:	1+2	=	2000 kg/12.6 m
D·	4+5+6+9	=	2135 kg/9.0 m
E:	8+3+7+10	=	2195 kg/9.5 m.

TABLE 7. 1985 PAYLOAD SET

No.	Mission	Mass (kg)	Length (m)	Diameter (m)
1	INTELSAT V	1000	6.3	2.6
2	INTELSAT V	1000	6.3	2.6
3	Other U.S.	475	3.5	2.2
4	TDRSS	1020	5.6	2.8
5	PALAPA	285	3.4	1.9
6	PALAPA	285	3.4	1.9
7	Foreign Communications	475	3.5	2.2
8	INATSAT	930	6.0	2.8
9	TELESAT D	545	0.9	3.7
10	GOES	315	3.1	1 9

Two conditions must now be satisfied First, the payload stacks must be matched to an IUS configuration capable of transporting them from the Shuttle to the IUS/Ion Drive exchange orbit. Second, the total mass and length of the IUS/payload stack combination must not exceed Shuttle cargo limits (29,500 kg mass, 18.3 meters length)

The two-stage IUS can deliver 3450 kg (plus 350 kg of mercury propellant for the Ion Drive stage) to the exchange orbit. Therefore, the first condition is satisfied for all five payload stacks However, when the IUS length of 4 5 meters is added to payload stacks A and B, the Shuttle length constraint is exceeded. Stacks C, D, and E do not violate this constraint. These results are summarized in Table 8.

TABLE 8. SHUTTLE/PAYLOAD STACK COMPATIBILITY

Payload Stack	Stack Length (m)	IUS Length (m)	Total (m)	Shuttle Length (m)	Length Limit Exceeded ?
A	16.9	4.5	21.4	18.3	Yes
B	15.1	4.5	19.6	18.3	Yes
C	12.6	4.5	17.1	18.3	No
D	9.0	4.5	13.5	18.3	No
E	9.5	4.5	14.0	18.3	No

Therefore, the IUS/Ion Drive propulsion option requires that three payload stacks and flights be used to transport all of the 1985 payloads. However, the mass of each of those payload stacks is 2200 kg or less. The two-stage IUS can deliver 2200 kg all the way to geosynchronous orbit. Thus, the use of the Ion Drive system is not required.

The conclusion that may be drawn from the foregoing analysis is that the combination of the payload definitions for the 1985 mission set and the Shuttle cargo bay length limit renders the low-thrust system ineffective when compared to straightforward IUS delivery.

Now, the 1985 payloads are all defined as being "dual compatible" They are all significantly longer (2-5 meters) than the "Shuttle optimized" payloads that appear in later years in the model (see Table 6). To determine the extent of improvement (from the low-thrust standpoint) that might be expected if Shuttle optimized payloads become the norm, a second analysis was conducted in which all payloads were assumed to be Shuttle optimized.

An arbitrary set of 33 payloads defined as SSUS-A' class in Table 6 was selected for the analysis.* These payloads have a mass of 1020 kg, are 1.5 m long, and are 4.3 m in diameter. The first of the 33 payloads was assumed launched with the Ion Drive when it is placed in the ~16,000 km exchange orbit. The remaining 32 payloads were assumed launched in eight stacks of four payloads each using the modified four-stage IUS. If it is assumed that the first payload can be mated to the first stack of four payloads arriving at the exchange orbit, then all 33 payloads can be delivered in eight Ion Drive trips to geosynchronous orbit, in a total time corresponding closely to the lifetime of the Ion Drive thrusters. Furthermore, all Shuttle cargo limits are satisfied.

The cost of this operation was calculated and compared to that for two chemical propulsion options: the two-stage IUS, and the Three-Stage IUS. The two-stage IUS can deliver two SSUS-A' payloads per trip, and the three-stage IUS can deliver three payloads per trip. Results are presented in Table 9.

The cost totals shown represent the total of all direct transportation charges: Shuttle, IUS stages, and Ion Drive stage. The differences in the totals are insignificant. The assumption of all Shuttle optimized payloads was sufficient to produce parity between Ion Drive and the chemical systems, but not sufficient to produce any significant saving using Ion Drive.

*The payload total of 33 was selected because it was convenient for costing the Ion Drive option. Under the chosen groundrules and assumptions, a single Ion Drive stage within its lifetime, can transport a total of 33 payloads to geosynchronous orbit.

TABLE 9. ALL SSUS A' VARIATION

Propulsion Option	IUS (2-Stage)	IUS (3-Stage)	IUS/Ion Drive (4-Stage IUS)
SSUS A' Payloads	1020 kg mass, 1.5 m long, 4.3 m diameter		
Total Payloads	33	33	33
Payloads/Launch	2	3	4
Total Cost ($M)	254.4	258.5	247.5
Ion Drive Assumptions	$27M unit cost, 8 trips/unit (20,000 hours thruster life) $27M development cost amortized over 100 payloads (10 years).		

If the Ion Drive (or the Sail) could be operated directly from the Shuttle with large payload stacks (up to eight payloads), then a significant transportation cost reduction (up to $100M) might occur. However, the large payload stacks would generate excessive trip times requiring significant increases in stage lifetime and probably incurring significant increases in mission associated costs (including spacecraft redesign for extended passage through the radiation belts).

In conclusion, it appears that chemical systems such as the IUS will be much better suited to the task of delivering small automated payloads to geosynchronous orbit through the decade of the 1980s. On the other hand, it should be noted that as new mission concepts evolve (as they should when the Shuttle becomes operational), a significant requirement for increased propulsion capabilities that could be met by low-thrust propulsion may emerge.

TECHNICAL AND OPERATIONAL CONSIDERATIONS

There are a number of technical and operational considerations which arise when considering the application of the Ion Drive and Solar Sail systems to Earth orbital missions. Some of these have been mentioned in previous discussions; others have not. The following paragraphs bring together and summarize the more significant issues.

Accessible Regions

Both systems are susceptible to damage from the Van Allen radiation belts. Ion Drive solar cells degrade very rapidly when first exposed to the radiation belts. The array power loss is approximately 50 percent during the first pass through the belts Subsequent trips produce significantly and progressively less damage due to "hardening" of the cells to the radiation flux. However, the initial damage greatly reduces performance and increases flight time, which reduces the number of trips possible within a given lifetime. The usefulness of the Ion Drive system in Earth orbit would benefit significantly from the introduction of solar cells that are relatively unaffected by Van Allen radiation However, presently, the cost penalty of doing so is prohibitive.

Sail performance is unaffected by the radiation, but the Sail material is gradually weakened to the failure point and the damage incurred is probably the life-limiting factor for the Sail in Earth orbit. In addition, the sail is more susceptible than the Ion Drive to two other Earth orbital environmental factors.

The Sail cannot operate directly from the Shuttle (as could the Ion Drive) because the aerodynamic drag in low orbits would exceed the Sail main propulsion force. The Sail must maintain an altitude of at least 1000-1500 kilometers.

Since the Sail is much larger than the Ion Drive, it is much more likely to be struck by other objects orbiting Earth. However, a hit on the Sail is less likely to be catastrophic than one on the Ion Drive.

Langley[9] estimates there is a 1 in 10 chance that the Sail will be hit in travelling to geosynchronous orbit However, JPL[2] estimates only a 5 in 1000 chance of a catastrophic hit.

On-Orbit Lifetime

On-orbit lifetime has a significant impact on propulsion system cost effectiveness. The relatively high cost of low-thrust systems can be more readily amortized if several missions can be performed with a single stage The Ion Drive thruster life of 15,000-20,000 hours is the limiting factor for the Ion Drive system. For most planetary missions, the propulsion module life is higher than for Earth orbital missions because most of the thrusters

may be shut down for significant periods of time. For Earth orbital missions all thrusters (except spares) operate continuously except during shadow periods.

Sail lifetime is probably limited by radiation damage to the Sail material. However, the Earth orbital debris hazard and the reliability of the Sail blade mechanical drive mechanisms may also be life-limiting factors. Sail lifetime in Earth orbit is probably comparable to that of Ion Drive.

Performance Sensitivity

Low-thrust performance is less affected by system degradation then are chemical systems. For chemical systems, degradation of system parameters is more likely to mean loss of capability to perform its intended mission. For low-thrust systems the primary effect of performance degradation is usually an increase in flight time. However, increased flight time has an economic impact through the reduced number of trips possible within a lifetime limit.

Thermal Cycling

As a result of Earth shadowing, thermal cycling is probably a more severe problem for Earth orbital missions than for planetary missions. Thermal gradients may cause distortions in solar arrays or Sail blades, causing performance degradation, control problems, and structural damage. The problem has been analyzed for flat solar arrays and found to be manageable. The introduction of concentrators could change that result

Attitude Control

The preliminary Earth orbital application of low-thrust propulsion may be orbit raising of large massive payloads. Control authority over large masses would be a problem for both the Sail and the Ion Drive--perhaps more so for the Ion Drive, because with present designs the entire propulsion system plus payload would have to be maneuvered for thrust vector pointing. The Sail may accomplish most of its thrust vector pointing from a relatively fixed attitude by cyclic pitching of its blades. However, the Sail may experience problems with gravity gradient disturbing torques.

Thermal Control

The Boeing PLUS study(4) concluded that low-thrust delivery of payloads to geosynchronous orbit gives rise to payload thermal control problems due to shadowing. To alleviate the problem, heaters must be added For the Ion Drive, a combination of solar array power and batteries would be used. For the Solar Sail, additional power would have to be added.

System Sharing

As larger geosynchronous payloads evolve, power requirements may grow significantly. Payloads placed in orbit by an Ion Drive system may be able to make use of the Ion Drive arrays to satisfy power requirements. Ion Drive systems could provide both propulsion and power for space solar power generation experiments.

Rendezvous and Docking

A rendezvous and docking capability will be a necessity if low-thrust systems must be reused several times to produce favorable economics. An Apollo-type probe and drogue system with man-in-the-loop control via TV could be designed and added to the Ion Drive system. However, for the Sail, the docking problem would be more difficult to solve. The Sail probably cannot assume an active docking role. Either the payload would have to dock with it, or a separable module might be needed to acquire the payload and bring it to the Sail. In any event, the Sail docking hardware would have to include a shock absorbing system to control docking dynamics.

CONCLUSIONS

There are a variety of potential future Earth orbital applications of low-thrust propulsion, including: payload delivery, payload servicing, technology verification, orbit debris control and manned mission support. For the decade of the 1980s, the most likely application is delivery of payloads to geosynchronous orbit. The other mission concepts (except technology verification as it applies to space solar power generation) are viewed primarily as longer term possibilities

Analyses show that low-thrust propulsion is not competitive with chemical systems for the delivery of the single or multiple small automated payloads that may be expected to dominate geosynchronous mission traffic in the 1980s. This is primarily due to the high cost of low-thrust propulsion systems as compared to conventional chemical systems. On the other hand, if a requirement for delivery of large payloads (e.g., large space antennas) to geosynchronous orbit evolves, then the use of low-thrust propulsion may prove desirable.

Present performance results indicate that the Ion Drive would produce better Earth orbital performance than the Solar Sail. This conclusion must be somewhat tempered by the knowledge that the Sail performance estimates are very preliminary and could improve under further analysis. However, Sail performance is not likely to exceed Ion Drive performance because of the impact of the Sail thrust vector pointing constraints. Those constraints are far more restrictive in Earth orbit than on the Halley mission, or planetary missions. Overall, the Ion Drive system appears to have better application potential for Earth orbital missions than the Solar Sail.

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(6) "Solar Sail System Definition Data", provided by J. L. Wright, NASA Jet Propulsion Laboratory, June, 1977.

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(9) Childress, O. S., "Solar Sail Review Team Final Report", NASA Langley Research Center, Final Report, June 15, 1977.

APPENDIX A

LOW-THRUST TRAJECTORY APPROXIMATION TECHNIQUE

A low-thrust trajectory approximation technique was used to generate the Ion Drive performance data required for this study. This Appendix describes the development of that technique.

In developing an approximation for a low-thrust trajectory, it is desirable to consider variations in orbital elements which change slowly. Starting with Lagrange's planetary equations for rates of change of semimajor axis and inclination:

$$\frac{da}{dt} = \frac{2a^2}{\sqrt{\mu p}} \left\{ F_r \, e \sin \theta + F_t \, (1+e \cos \theta) \right\} \quad , \qquad (1)$$

$$\frac{di}{dt} = \frac{r \, F_n}{\sqrt{\mu p}} \cos u \quad , \qquad (2)$$

where p is the semilatus rectum, e is the eccentricity of the orbit, F_r, F_t and F_n are the radial, transverse, and normal components of acceleration, θ is the true anomaly, and u is the argument of latitude.

The low-thrust system is assumed to operate between two circular orbits, for which the eccentricity is zero. Furthermore, the eccentricity is assumed to remain zero in transit. The components of acceleration will be taken as.

$$f_r = 0, \; F_t = \frac{T \cos \phi}{m_o - \dot{m}t} \; , \; F_n = \frac{-T \sin \phi}{m_o - \dot{m}t} \operatorname{sgn} (\cos u) \quad , \qquad (3)$$

where ϕ is an angle which represents the split of the thrust between altitude change and inclination change, m_o is the initial mass, $\dot{m}$ is the mass flow rate, and sgn is the sign function. The formulation is being developed for raising the orbit and reducing the inclination, but the final results will also apply to the return case. Substituting the components of acceleration into Lagrange's equations and letting the eccentricity be zero (which implies the semilatus rectum is equal to the semimajor axis), gives the following:

$$\frac{da}{dt} = \frac{2a^{3/2}}{\sqrt{\mu}} \frac{T \cos \phi}{m_o - \dot{m}t} \quad , \tag{4}$$

$$\frac{di}{dt} = - \sqrt{\frac{a}{\mu}} \; |\cos u| \; \frac{T \sin \phi}{m_o - \dot{m}t} \quad . \tag{5}$$

Separating variables in Equation (4) gives:

$$\frac{da}{a^{3/2}} = \frac{T \cos \phi}{m_o - \dot{m}t} \cdot \frac{2dt}{\sqrt{\mu}} \quad , \tag{6}$$

and integrating holding ϕ constant gives:

$$\frac{1}{\sqrt{a_f}} - \frac{1}{\sqrt{a_o}} = \frac{T \cos \phi}{\dot{m} \sqrt{\mu}} \log \left(1 - \dot{m}t_f/m_o\right) \quad , \tag{7}$$

where a_f and a_o are the final and initial values of semimajor axis and t_f is the final time. Typically, the initial and final altitudes are known, as well as the system parameters T, $\dot{m}$ and m_o; thus, the final time could be determined if the angle ϕ were known. In preparation for integrating Equation (5), the $\sqrt{a/\mu}$ as a function of time is given as:

$$\sqrt{\frac{a}{\mu}} = \left\{ \frac{T \cos \phi}{\dot{m}} \log \left(1 - \frac{\dot{m}t}{m_o}\right) + \sqrt{\frac{\mu}{a_o}} \right\}^{-1} \quad . \tag{8}$$

Substituting this result into Equation (5) gives·

$$\frac{di}{dt} = - \left\{ \frac{T \cos \phi}{m} \log \left(1 - \frac{\dot{m}t}{m_o}\right) + \sqrt{\frac{\mu}{a_o}} \right\}^{-1} |\cos u| \frac{T \sin \phi}{m_o - \dot{m}t} . \quad (9)$$

This equation can be integrated in closed form if the $|\cos u|$ could be represented by a constant, $1/K$. The average value of $|\cos u|$ is $2/\pi$, which would correspond to changing the inclination all around the orbit. The more optimal strategy would be to do the inclination change at the nodes only where $|\cos u|$ is 1. The actual choice of the constant will be discussed with the evaluation of the other constants. Letting $|\cos u|$ be $1/K$ and $x = \log (1-\dot{m}t/m_O)$, we have, by integration:

$$\frac{K\dot{m}\Delta i}{T \sin \phi} = \int \frac{dx}{\frac{T}{\dot{m}} \cos \phi \; x + \sqrt{\frac{\mu}{a_o}}} , \quad (10)$$

$$= \frac{\dot{m}}{T \cos \phi} \log \left(\frac{T \cos \phi}{m} x + \sqrt{\frac{\mu}{a_o}} \right) \Bigg|_{x=0}^{x=\log \left(1-\frac{\dot{m}t_f}{m_o}\right)} . \quad (11)$$

To simplify Equation (11) and use terminology consistent with low-thrust systems, the following relationships are used.

$$T = \dot{m}c, \; p_J = \frac{1}{2} \dot{m} c^2, \; v = \sqrt{\frac{\mu}{a}} , \quad (12)$$

where c is the jet velocity, p_J is the jet power, and v is the equivalent circular orbit velocity; additionally, at $t = t_f$, from Equation (8)

$$\frac{I}{\dot{m}} \cos \phi \log (1 - \dot{m}t_f/m_o) + \sqrt{\frac{\mu}{a_o}} = v_f . \quad (13)$$

Thus, thc angle ϕ can be determined from:

$$\tan \phi = \frac{K \Delta i}{\log (v_f/v_o)} . \quad (14)$$

For the upbound leg, Δi is negative, v_f is less than v_o, and ϕ is between 0 and 90 deg, for the down leg, Δi is positive, v_f is greater than v_o, and ϕ is between 180 and 270 deg. However, in both cases the same equations are valid. Solving Equation (7) for t_f and substituting the relationships in Equation (12) gives.

$$t_f = \frac{m_o c^2}{2p_J}\left(1 - e^{\frac{v_f - v_o}{c \cdot \cos\phi}}\right) \approx \frac{m_o c}{2p_J}\frac{(v_o - v_f)}{\cos\phi} \quad . \tag{15}$$

These last two equations provide a method for estimating performance to and from geosynchronous orbit with a low-thrust system once a value of K is chosen.

Several assumptions have been made in the development of these approximations. These have been examined by comparing the results of these approximations with data generated by MSFC.* The key assumptions are:

(1) The eccentricity remains zero.

(2) The rate of change of semimajor axis and inclination are approximately proportional (i.e., ϕ is constant).

(3) The radiation belts are not considered.

(4) K is chosen as the average of the two extremes (K = 1.2854).

(5) The transfers are between circular orbits.

Due to Assumption (3), data were checked only for cases completely above the radiation belts. The results and the various assumptions were found to hold reasonably well; the eccentricity remained small, holding ϕ constant is a valid assumption, and the estimates of the transfer times agreed within a few percent.

The following method has been developed to extend the procedure to trajectories which traverse the radiation belts. A radiation flux model and solar cell damage model were obtained from MSFC. The major effect of the radiation is to alter the thrust. Thus, Equations (4) and (5) can be numerically integrated, with the thrust being evaluated from the integrated flux and the radiation damage model. The choice of ϕ is obtained from Equation (14). By replacing $|\cos u|$ with a constant factor, the numerical integration did not have to be done at steps commensurate with the orbital motion, but rather several days per step. Although the trajectories from low Earth orbit to geosynchronous do not remain circular, the final time estimates agreed well with data from MSFC. The obvious advantage of this procedure is that it enables data and trade-offs of various parameters to be obtained without requiring the lengthy computer runs needed for converged trajectories from programs such as SECKSPOT or MOLTOP. Those programs, however, are required to evaluate how accurate the approximations are.

*SEPS Performance Analysis Data, obtained from C. Russell, NASA Marshall Space Flight Center, 1977.

APPENDIX B

1985-1990 OUTSIDE USER GEOSYNCHRONOUS MISSION MODEL

A 1985-1990 outside user's geosynchronous mission model was generated for use in assessing low-thrust mission capabilities. The model is summarized in Table B-1. The information presented in Table B-1 was derived from Battelle's August 1977 Revised Outside Users Payload Model*, which consists of two payload model variants. The variants, termed "high" and "low", represent roughly ± 2 sigma projections of future activity of non-NASA, non-DoD payloads. Table B-1 lists geosynchronous payloads likely to fly on the STS during the six-year time period from 1985 to 1990. Launch schedules associated with each payload do not exactly reflect either the high or low models, but fall within their bounds

*Neale, D. B , "Outside Users Payload Model", BMI-NLVP-IM-77-4, August 15, 1977.

TABLE B-1. 1985-1990 OUTSIDE USER GFOSYNCHRONOUS MISSION MODEL

Mission Name	Launch Schedule						Mass		Length		Diameter	Load Factor	SSUS Class
	85	86	87	88	89	90	With AKM	Without AKM	S/C	S/C+ U.S.		S/C + U.S.	
INTELSAT V	2	1					2000	1000	6.3	8.7	2.6	0.63	A
INTELSAT VI					1	2	2040	1020	1.5	3.9	4.3	0.28	A'
COMSTAR						2	1910	955	7.2	9.6	2.6	0.70	A
RCA					1	2	1090	545	0.9	3.0	3.7	0.22	D'
MARISAT			2	1			660	330	3.8	5.9	2.2	0.43	D
SBS			2	1			2040	1020	1.5	3.9	4.3	0.28	A'
Public Service					1		2040	1020	1.5	3.9	4.3	0.28	A'
Other U.S.	1	1	1	1	1	1	950	475	3.5	5.6	2.2	0.41	D
Disaster Warning					1	1	1090	545	0.9	3.0	3.7	0.22	D'
TDRSS/WESTAR	1			2			2040	1020	5.6	10.4	2.8	0.76	A
Palapa	2	1					570	285	3.4	5.5	1.9	0.40	D
Brazil		1	1				950	475	3.5	5.6	2.2	0.41	D
Other Foreign Comm.	1	1	1	1	1	1	950	475	3.5	5.6	2.2	0.41	D
INATSAT	1	1	2	1			1860	930	6.0	8.4	2.8	0.61	A
TELESAT D	1						1090	545	0.9	3.0	3.7	0.22	D'
TELESAT				1	1	1	1090	545	0.9	3.0	3.7	0.22	D'
UHF				1	1		1090	545	0.9	3.0	3.7	0.22	D'
GOES	1	1			1	1	630	315	3.1	5.2	1.9	0.38	D
INMETSAT		1	1		1	1	950	475	3.5	5.6	2.2	0.41	D
E.R.(U.S. Govt.)						1	1090	545	0.9	3.0	3.7	0.22	D'